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AN INVESTIGATION OF THE AERODYNAMIC CHARACTERISTICS  
OF A NEW GENERAL AVIATION AIRFOIL IN FLIGHT

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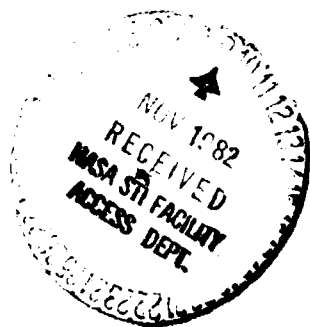
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Langley Research Center  
Hampton, Virginia 23665

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**Aeronautical and  
Astronautical Research  
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**AN INVESTIGATION OF THE AERODYNAMIC CHARACTERISTICS  
OF A NEW GENERAL AVIATION AIRFOIL IN FLIGHT**

**G. M. GREGOREK, M. J. HOFFMANN AND G. S. WEISLOGEL**

**FINAL REPORT TO  
THE LANGLEY RESEARCH CENTER  
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION**

**GRANT NO. NSG-1184**

**OCTOBER 1982**

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IN-FLIGHT MEASUREMENTS OF THE GA(W)-2 AERODYNAMIC  
CHARACTERISTICS

## I. INTRODUCTION

In May 1975, The Ohio State University (OSU) was awarded Grant No. NSG-1184 from the Langley Research Center of the National Aeronautics and Space Administration. Entitled "An Investigation of the Aerodynamic Characteristics of a New General Aviation Airfoil in Flight," the purpose of the grant was to evaluate the second of a new series of low speed airfoils designed by personnel of the Langley Research Center - the GA(W)-2\* - a 13% thickness to chord ratio airfoil. Beech Aircraft Corporation (BAC) was to be an active participant in the research program, providing the aircraft fabricating the airfoil to be "gloved" onto the wing, and assisting in the engineering and data analysis.

Accordingly, the wing of a Beech Sundowner was modified at BAC by adding balsa ribs and covered with aluminum skin, to alter the existing airfoil shape to that of the GA(W)-2 airfoil. The aircraft was flown in a flight test program that gathered wing surface pressures and wake data from which the lift drag, and pitching moment of the airfoil could be determined. After the base-line performance of the airfoil was measured, the drag due to surface irregularities such as steps, rivets and surface waviness was determined. The potential reduction of drag through the use of surface coatings such as KAPTON was also investigated.

The last flight in the 46 flight test program was flown February 21, 1977 and the aircraft returned to Beech soon after. Four technical papers were written describing the results of this flight research; the purpose of this report is to summarize this work. The technical papers are included as an Appendix.

## II. RESEARCH RESULTS

### General Approach

As noted earlier, the main objective of the research was the flight evaluation of the GA(W)-2 airfoil. While moving toward this objective, several research methodologies were developed that were of interest to the technical community. These developments, associated with the aircraft modifications and flight procedures, the instrumentation used to make the measurements, and the surface pressure and wake measurements themselves, were reported in a timely manner at national technical meetings. The three major areas are touched upon below.

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\*The early airfoil designation which stood for General Aviation airfoil, Whitcomb design number 2 has been replaced by the designation LS(1)-0413 for Low Speed series 1 airfoil with design lift coefficient 0.4 and thickness to chord ratio of 0.13.



## Flight Test Aircraft

The modification to the wing of the Beech Model C23 Sundowner was worthy of note. Although airfoil "gloves" have been used in the past, the simple balsa ribs with bonded aluminum skin used by Beech Aircraft Corporation to construct the GA(W)-2 airfoil section over the standard NACA 63415 airfoil was unique, forming a light-weight, smooth contour that produced the GA(W)-2 section accurately and at low cost.

A special problem existed with the glove because the thickness to chord ratio of the GA(W)-2 is 0.13, less than the 0.15 thickness ratio of the normal wing. The balsa ribs extended the leading edge by seven inches and the trailing edge by two inches. Because of these changes Beech Aircraft Corporation recommended new operating limitations for the test aircraft:

	Model C23 Normal Category	Model C23 GA(W)-2 Wing
Never Exceed Speed	175 mph	120 mph
Maximum Positive Load Factor	3.8 G	2.0 G
CG Range	10.5 inch	2.0 inch

### Flight Into Visible Moisture To Be Avoided

The gloved airfoil did not affect the handling qualities of the aircraft in any significant fashion, although the aft loaded airfoil required increased cable tension on the ailerons to minimize up-float due to large aileron hinge moments.

The aircraft was an excellent platform with which to evaluate the airfoil in flight. Additional details of the modifications and operational procedures used to obtain the data are presented in the Appendix in "GA(W)-2 Airfoil Flight Test Evaluation" SAE PAPER No. 760492 presented at the Business Aircraft Meeting, April 1976.

## Instrumentation

To evaluate the airfoil, detailed measurements of the airfoil surface pressures and the total and static pressure distribution in the wake behind the wing were required. Integration of the surface pressure along the chord of the airfoil yields the lift and moment coefficient while integration of the momentum loss across the wake obtained from the total and static pressures provides the drag coefficient. By obtaining this distribution at different flight angles of attack, the section drag polar is built up.

The airfoil surface pressures were measured by a belt of plastic tubing wrapped chordwise about the wing, each tube provided with a single orifice to measure the surface pressure at a

single chordwise station. Forty such tubes led to a somewhat unusual pressure sensing apparatus, one that used a cut-off valve to trap the sensed pressures prior to measuring them with a scanning valve and single transducer. With this innovation, all surface pressures were caught at one instant in the flight profile, allowing the trapped pressures to be scanned and recorded on FM tape at a leisurely pace, minimizing airfoil pressure variations due to any flight path disturbances. The quality of the surface pressure data obtained by this arrangement was excellent.

The wake surveys were measured with a rotating probe equipped with separate total and static pressure probes. Probe angular position was determined from a potentiometer which could then be converted to probe position above and below the airfoil chord line. With sensitive pressure transducers and accurate position data drag coefficients were measured to within  $\pm 2$  drag counts.

The transducer signals as well as air speed and angle of attack information were recorded on FM tape at selected air speeds during a series runs in level flight. After landing, the tape was processed by computer using efficient software developed for this flight research program. A typical flight with 17 different air speeds and several thousand pressure, position and angle measurements could be reduced to finished plots within three hours of touchdown.

The pressure measuring and data acquisition systems and the flight test techniques developed for airfoil evaluation during this grant are described in more detail in the Appendix in two papers: "Data Acquisition System for In-Flight Airfoil Evaluation" SAE Paper No. 760462 presented at the Business Aircraft Meeting in April 1976 and in "Flight Test Techniques for Low Speed Airfoil Evaluation" presented at the Advanced Technology Airfoil Research Conference held at NASA Langley Research Center in March 1978 and published in the Proceedings in NASA Conference Publication 2045.

### Flight Test Results

The flight experiments verified the basic concepts of the research program; i.e., the gloved airfoil technique was an economical means to acquire airfoil information in flight, and the pressure measuring systems were reliable, producing data of excellent quality from which the airfoil lift, drag, and pitching moment could be resolved. With this airfoil evaluation methodology, several results of importance were obtained.

Drag polars of the GA(W)-2 airfoil were measured for different configurations; with the airfoil clean, with leading edge grit to trip the boundary layer at the leading edge and at the aileron station. The flight data verified the performance of the airfoil measured in the wind tunnel and showed the penalties of early transition and the large drag increase at the aileron station.

Detailed surface pressures were obtained at five spanwise stations on the wing. When these pressures were compared with analytic predictions, excellent agreement between theory and flight pressure distributions were noted.

Surface irregularities were examined by building forward facing and aft facing steps on the airfoil, fabricating surface waves on the upper and lower surface of the airfoil and placing rows of button-head rivets on the airfoil surface. Drag polars taken with these surface imperfections provide design information of the drag penalty associated with these fabrication techniques. For example, backward facing steps produced as much drag increment as forward facing steps, surface waves can act as an effective trip to the airfoil boundary layer and the lower surface of the GA(W)-2 (and all airfoils in general) should be kept as clean as possible to take advantage of the potential for laminar flow that exists at modest angles of attack.

A significant flight test with a KAPTON surface coating - a polyimide fiber of 0.005" thickness resembling mylar - produced a surprising reduction in drag, opening up a new research technique for drag reduction.

Results of these flight tests and selected comparisons with theory and with wind tunnel measurements are given in the Appendix in "In-Flight Measurements of the GA(W)-2 Aerodynamic Characteristics," SAE Paper No. 770461 presented at the April 1977 Business Aircraft Meeting in Wichita, Kansas.

### III. SUMMARY

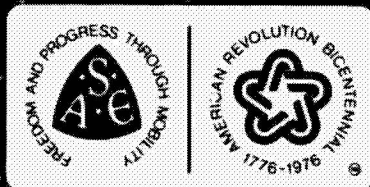
The results of an evaluation of the GA(W)-2 airfoil supported by NASA Langley Research Center Grant NSG 1184 have been summarized in this report with the Appendix providing the detailed discussions. The basic result is that the GA(W)-2 airfoil (now designated the NASA LS(1)-0413) will perform as measured in the wind tunnel and as predicted by analytic methods.

With an economical wing contour "gloved" to the existing wing, the Beech Sundowner equipped with pressure belts, wake survey probe and a modern flight data acquisition package measured drag polars, surface pressure distributions - both chordwise and spanwise - for a variety of flight conditions. Flights aimed at measuring airfoil sensitivity to manufacturing tolerances provided data with which to estimate drag penalties that accrue to steps and roughness elements. An interesting flight sequence with film coatings showed reduced drag due to the smooth surface films that warrants further study.

The research effort also demonstrated the usefulness of the analytic tools now available for airfoil and wing design. Pressure distributions, incidence angle corrections for finite wing effects and drag predictions were predicted analytically and shown to agree very well with the flight results.

These results have been passed on to the aviation community to improve the technology available for aircraft design.

## APPENDICES



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## **GA(W)-2 Airfoil Flight Test Evaluation**

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GA(W)-2 AIRFOIL  
FLIGHT TEST EVALUATION

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ABSTRACT

A brief description of the GA(W)-2 Airfoil Flight Test Evaluation Program is presented. Employing an economical approach to airfoil flight testing, the GA(W)-2 airfoil was "gloved" on the existing wing structure of a Beech Model C23 "Sundowner". Program objectives, experimental approach, research aircraft modification and instrumentation, data acquisition and processing, flight operations, and preliminary flight test results are described.

INTRODUCTION

In recent years the National Aeronautics and Space Administration has refocussed its research efforts on aeronautical problems. Of particular interest to the general aviation community is the airfoil research program (1). Initiated more than a decade ago to explore airfoils that could operate efficiently near the speed of sound, this airfoil research led to the supercritical airfoils now being tested on prototype aircraft.

The design technique developed as a result of this recent research - flow analyses and airfoil design supported by extensive computer studies and wind tunnel tests - are now being used to develop a new series of low speed airfoils. The first of these was the GA(W)-1, (for General Aviation, Whitcomb Airfoil Number 1) a 17% thick airfoil with a design lift coefficient of 0.4. This airfoil is being evaluated on the modified Piper Seneca in the Advanced Technology Light Twin (ATLIT) program (2). The second of the new low speed series is the 13% thick GA(W)-2 airfoil. This promising airfoil, also with a design lift coefficient of 0.4, emerged from the NASA Langley Research Center in June 1975. It is now being flight tested on a Beech Sundowner at The Ohio State University to confirm the potential benefits of the design.

Potential benefits of the GA(W)-2 airfoil design are derived from

its higher lift to drag ratio ( $L/D$ ) and maximum coefficient of lift ( $C_{l_{max}}$ ). Higher  $L/D$  tradeoffs include: (a) faster cruise speed, (b) increased range (lower fuel consumption), and (c) increased gross weight. Higher  $C_{l_{max}}$  tradeoffs include: (a) slower stall speed, (b) smoother ride, and (c) increased rate of climb.

The purpose of this paper is to describe the GA(W)-2 Airfoil Flight Test Evaluation now underway at The Ohio State University. This research is supported by NASA Langley Research Center Grant NSG-1184. Beech Aircraft Corporation is an active participant in the investigation, providing the aircraft, fabricating the airfoil glove, and assisting in the analysis of the data and preparation of the final report.

#### PROGRAM OBJECTIVES

The objectives of the GA(W)-2 Flight Test Evaluation Program are to:

1. Demonstrate the aerodynamic efficiencies of the GA(W)-2 airfoil.
2. Confirm theoretical and wind tunnel results.
3. Demonstrate a low cost approach to airfoil flight testing.

#### EXPERIMENTAL APPROACH

The experimental approach, designed to provide extensive in-flight aerodynamic data on the GA(W)-2 airfoil, contains four elements:

1. Surface pressure distributions, to derive airfoil lift characteristics.
2. Wake surveys, to derive airfoil drag characteristics.
3. Boundary layer surveys, for comparison to analytical predictions and wind tunnel tests.
4. Flight demonstration of the airfoil.

The flight test program, currently underway, will provide airplane lift coefficient vs. angle of attack, airfoil section lift and drag coefficient at two spanwise stations, surface pressure distributions at five spanwise stations, section moment coefficient vs. angle of attack, and boundary layer profiles vs. chord. The pressure distribution obtained from flush pressure taps will be compared to a distribution obtained from a pressure belt at the same spanwise location to identify any flow disturbances caused by the belt, and demonstrate an economical approach to acquiring airfoil pressure distribution data.



## RESEARCH AIRCRAFT MODIFICATION

A Model C23 "Sundowner 180" (Figure 1), a single engine, four place airplane, was used as the research vehicle for the flight test program. Owned by Beech Aircraft Corporation, it was modified by Beech under subcontract to The Ohio State University. An economical approach to airfoil flight testing was used. The GA(W)-2 airfoil shape was built up around the existing 63<sub>2</sub>A415 wing structure. The buildup consisted of balsa wood ribs and formers bonded to the exterior of the C23 wing. Aluminum skin bonded to the ribs and formers were contoured to the NASA supplied airfoil coordinates (Figure 2). The resulting "glove" has the desired airfoil characteristics, obtained without altering the existing wing structural integrity. Since the thickness to chord ratio of the Model C23 airfoil is 0.15, it was necessary to extend the leading edge seven inches and the trailing edge two inches in order to obtain the 13% GA(W)-2 airfoil shape. Each wing contains an access channel on the lower surface for pressure tubing. The left wing has ten flush pressure taps, six taps on the upper surface and four on the lower surface.

Beech performed an analysis of the wing modification to assure its structural integrity and conducted initial flights to assure the airworthiness of the modified airplane.

The modified airplane is subject to the following recommended operating limitations:

	Model C23 <u>(Normal Category)</u>	Model C23 Modified With <u>GA(W)-2 "Glove"</u>
Never Exceed Speed	175 mph	120 mph
Maximum Positive Load Factor	3.8 G	2.0 G
C.G. range	10.5 in.	2.0 in.

Flight in visible moisture is to be avoided. The modified airplane has no flaps.

The gloved airfoil does not affect the airplane handling qualities in any significant fashion. The airplane has proven to be an excellent platform from which to conduct an evaluation of the GA(W)-2 airfoil in its true environment.

## RESEARCH AIRCRAFT INSTRUMENTATION

The test aircraft has the normal complement of flight instruments supplemented by instrumentation to accurately measure airspeed and attitude from booms mounted under each wing tip. The left boom

supports a pitot-static probe mounted on a swivel to keep it aligned with the relative wind. Total pressure from the probe is used as the reference pressure for the pressure sensing system. The right boom houses the angle of attack sensor.

Figure 3 illustrates the pressure sensing instrumentation. Surface pressure distributions are measured using strip-a-tube belts taped around the left wing. Two belts, each consisting of ten 0.2" diameter plastic tubes plugged near their midpoints, provide forty chordwise surface pressures at a single spanwise station. Five spanwise stations have been examined to determine the complete pressure field on the wing.

Mounted on a rotating shaft attached to a drive mechanism under the right wing, a wake survey probe samples static and total pressure vertically in the airfoil wake inboard and outboard of the aileron gap. A ten tube boundary layer probe is used to measure the boundary layer profiles at different chord locations at a single spanwise station.

Tubing from the surface pressure belt, wake survey and boundary layer probes lead into the cabin through an access channel in each wing. In the cabin, two cutoff valves accept the pressure leads and couple them to a 48 port pressure scanning system. All pressures are measured by a pressure transducer and are recorded on magnetic tape as the scan valve steps through the trapped volume between the cutoff valves and the transducer.

#### DATA ACQUISITION AND PROCESSING

Flight test data is recorded by a seven channel FM magnetic data tape recorder on board the research aircraft. The basic elements of the data processing system are shown in Figure 4. After a test flight the tape is played back in the computer facility at The Aeronautical and Astronautical Research Laboratory, located on The Ohio State University Airport. The analog signal is digitized and stored on magnetic tape. It is then reviewed on a CRT display, and if the run has been normal, the digitized pressure belt data are processed to surface pressure coefficient form and integrated for lift and pitching moment, plotted and printed out in hard copy form. The wake data is similarly reduced to drag coefficient and presented in a plotted and a printed format.

The data processing is fully automated, with the exception of the CRT review process, which is maintained to assure data quality. In operation, the instrumentation and data processing systems have accepted the tape from a single 90 minute flight consisting of seventeen level runs and processed the data in less than three hours.

The final data was in the form of seventeen surface pressure distributions, 68 wake distributions (four surveys were taken at each flight speed), in finished plotted form with printouts of pertinent information. A more detailed description of the instrumentation and data acquisition system is contained in Reference 3.

## FLIGHT OPERATIONS

All research flights originate from The Ohio State University Airport, a major general aviation facility located in northwest Columbus.

A typical data flight lasting eighty minutes consists of level flight at 5000 feet altitude at indicated airspeeds ranging from 120 mph to 60 mph in 5 mph increments. Airspeeds are maintained within  $\pm 1$  mph and altitude within  $\pm 25$  feet during data recording. Corresponding aircraft lift coefficients vary from 0.4 to 1.6 with Reynolds numbers of the order of  $4 \times 10^6$  down to  $2 \times 10^6$ , based on wing chord.

Flights are conducted in accordance with a Flight Operations Plan prepared by The Ohio State University Department of Aviation. The Plan includes provisions for aircraft airworthiness, applicable Federal Aviation Regulations, Federal Aviation Administration coordination, named flight crew, flight test area designation, flight descriptions, aircraft handling and maintenance, operational safety, and insurance.

The Department of Aviation Aircraft Maintenance Division provides an extensive and experienced capability in support of the flight test program.

## PRELIMINARY FLIGHT TEST RESULTS

A mid-span surface pressure distribution is shown in Figure 5. The agreement of the measured flight data with theory (4) is evident. This is typical of the pressure distributions obtained during the flight program.

A typical wake survey is shown in Figure 6. Integrations of the wake survey and belt data at the same spanwise station allow comparisons of the GA(W)-2 section aerodynamic characteristics to be made. A comparison of the lift-drag polars for flight and wind tunnel is shown in Figure 7.\* Airfoil lift to drag ratio for the flight and wind tunnel cases are shown in Figure 8.

---

\* These curves are shown without scales because of the restriction on the release of preliminary data on the new airfoil at this time.

## SUMMARY

A brief description of the research to obtain a flight evaluation of one of a new generation of low speed airfoils has been presented. The research has been a cooperative effort between Beech Aircraft Corporation, NASA Langley Research Center, and The Ohio State University. This government-industry-university team assembled a sensitive pressure sensing system, placed it on board a single engine general aviation aircraft equipped with an economical modification of the GA(W)-2 airfoil, and used a sophisticated data acquisition and processing system to evaluate the performance of the new airfoil. Preliminary results compare favorably with both wind tunnel and theoretical evaluations. This work indicates that the promise of improved performance with the GA(W)-2 airfoil can be achieved.

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Figure 1. Beech Model C23 "Sundowner 180" Research Aircraft  
Modified with GA(W)-2 Airfoil

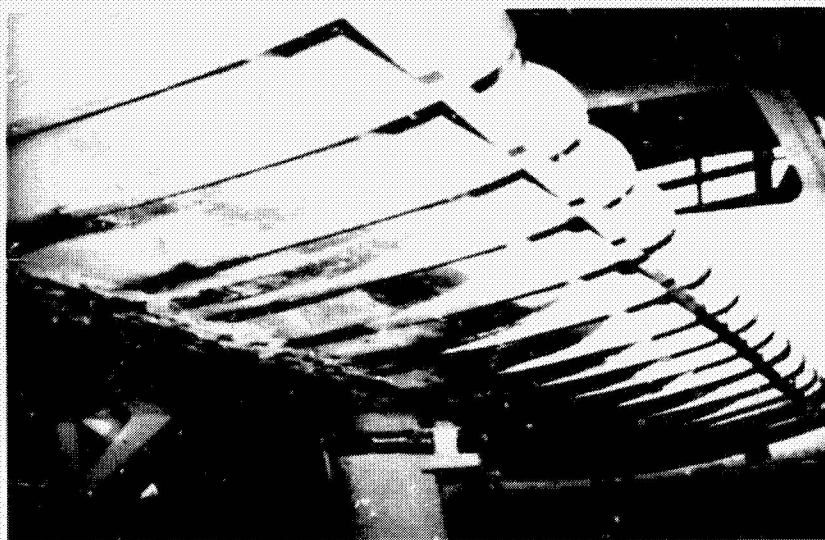


Figure 2. Fabrication of GA(W)-2 Airfoil "Glove"  
on Existing 63, A415 Wing

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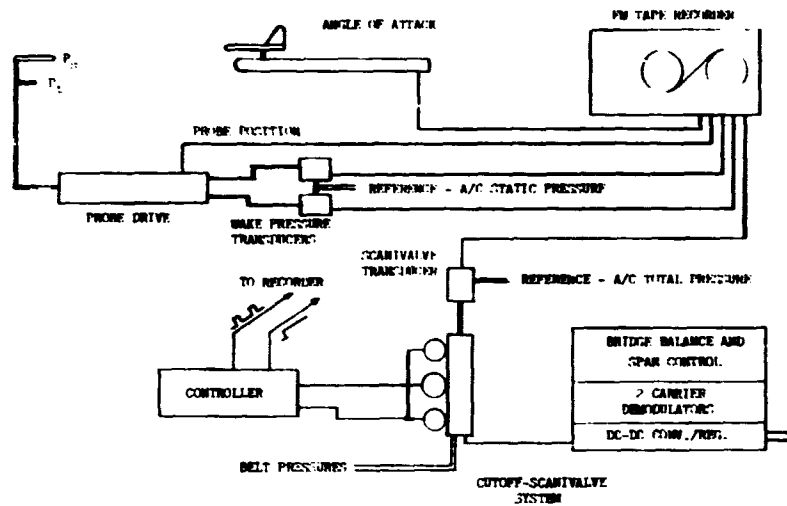


Figure 3. Pressure Sensing Instrumentation Schematic

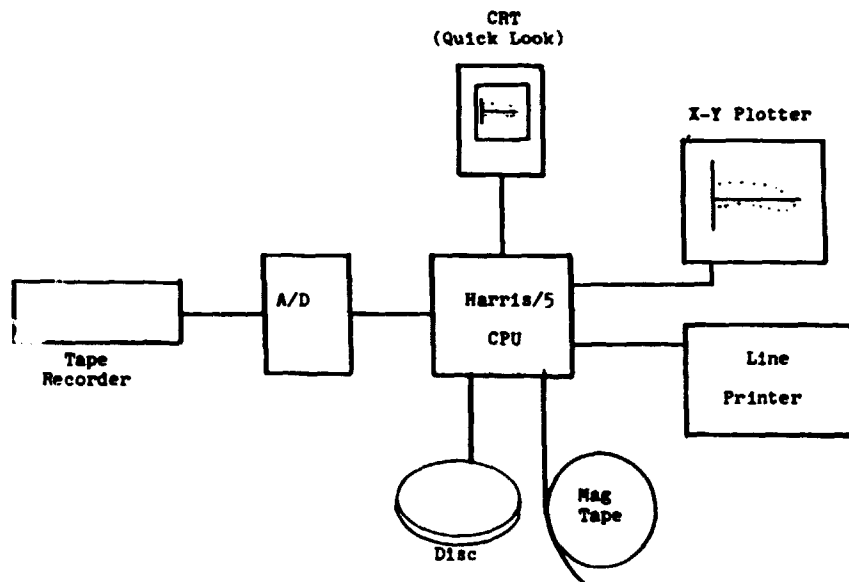


Figure 4. Data Processing Schematic

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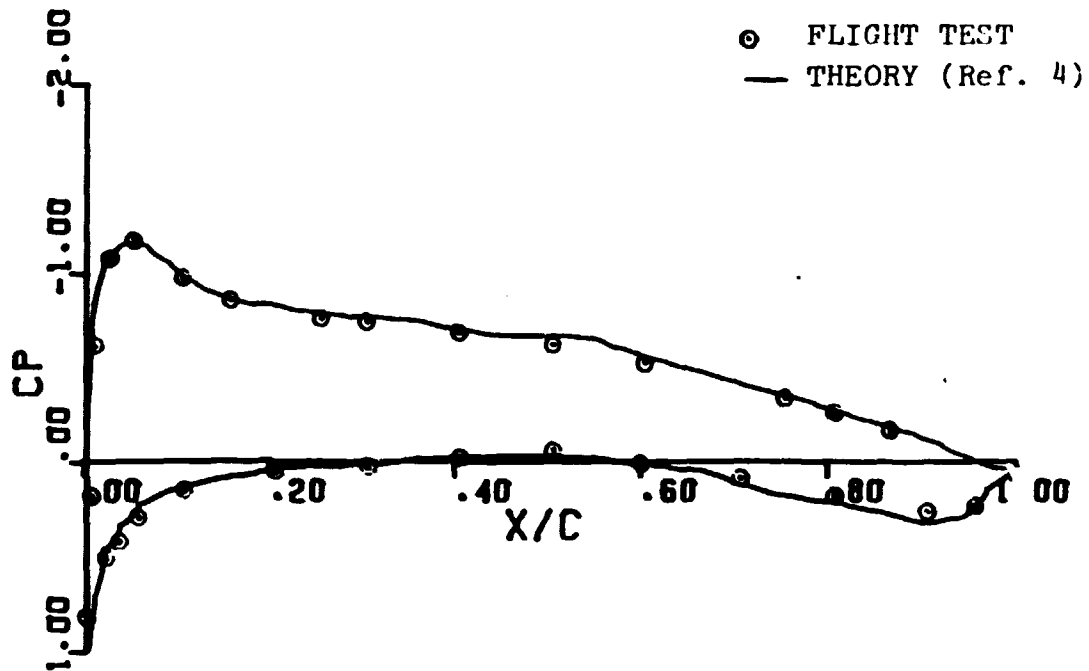


Figure 5. Comparison of an Experimental Pressure Distribution with Theory

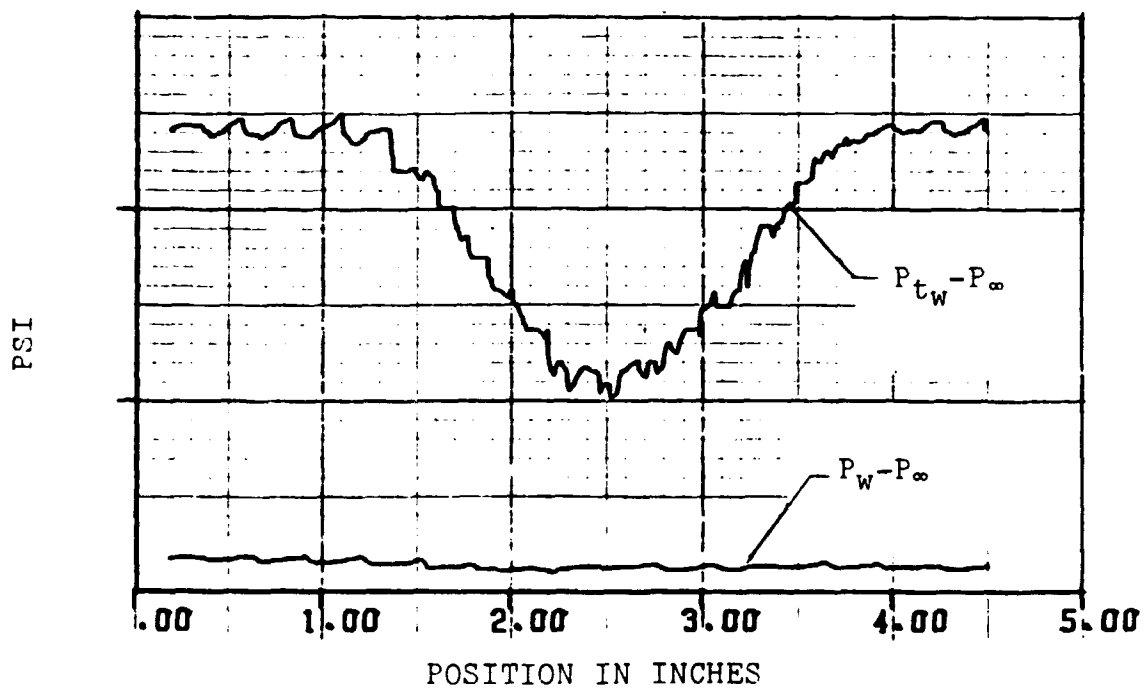


Figure 6. Typical Wake Survey

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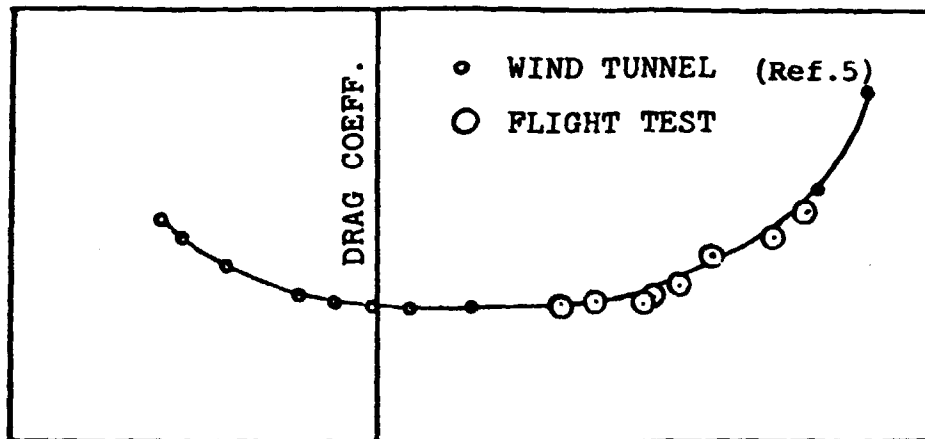


Figure 7. Preliminary Comparison of Drag Polars from Wind Tunnel and Flight Test

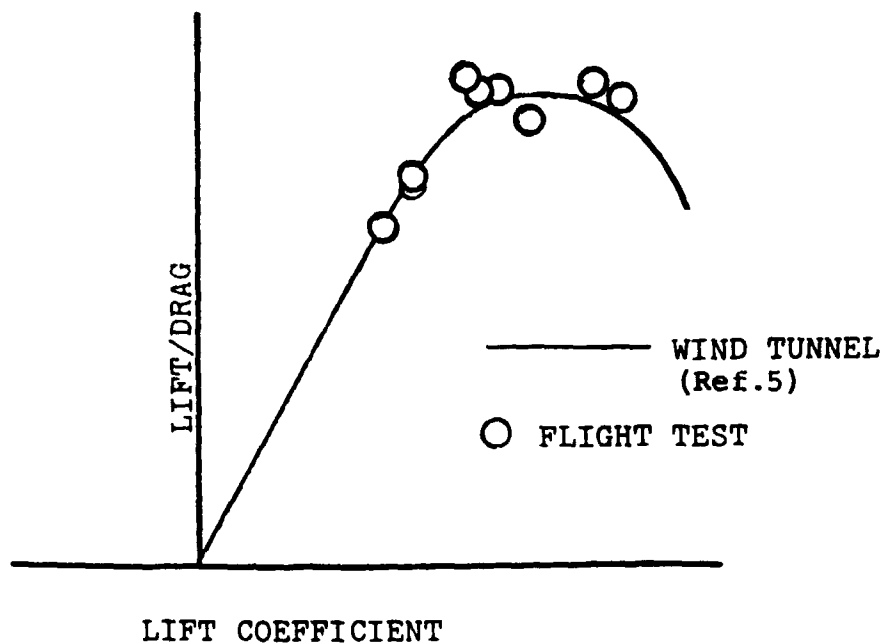
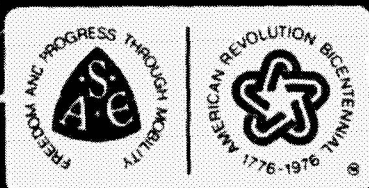


Figure 8. Preliminary Comparison of Lift to Drag Ratio from Wind Tunnel and Flight Test





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## **Data Acquisition System for In-Flight Airfoil Evaluation**

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A DATA ACQUISITION SYSTEM  
FOR IN-FLIGHT AIRFOIL EVALUATION

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#### ABSTRACT

Details of the design and development of an airborne data acquisition system for in-flight evaluation of airfoils are presented. The system was designed to be flown aboard a single engine general aviation aircraft and to measure and record airfoil surface pressures, airfoil wake pressures, and aircraft angle of attack and airspeed. Included are descriptions of the instrumentation, calibration and data reduction techniques, illustrations of the raw data and comments on the operational experience gained during the flight evaluation of the GA(W)-2 airfoil.

#### INTRODUCTION

Pressure measurements in-flight have always been of concern to flight test engineers. Measuring techniques have varied from use of simple water manometers to use of sophisticated multiple pressure transducer-scanning systems. Under NASA Langley Research Center Grant NSG 1184, entitled "An Investigation of the Aerodynamic Characteristics of a New General Aviation Airfoil in Flight", personnel at The Aeronautical and Astronautical Research Laboratory (AARL) have had the opportunity to design, build, and operate a sensing system for in-flight pressure measurements. The requirements of the instrument package were to measure and to record airfoil surface pressures, wake static and total pressures, wake probe position and aircraft attitude and airspeed. Since the acquisition system was to be flown on a single engine, general aviation aircraft, overall size, weight, portability and compatibility with the aircraft environment were additional design constraints.

The purpose of this paper is to detail the development of this flight data acquisition system and to relate the experience of the AARL staff in the operation of this airborne pressure sensing system on the flight evaluation of the GA(W)-2 airfoil (1).

## INITIAL CONSIDERATIONS

In the design stage, two major considerations contributed to the configuration of the data acquisition system. The first was the necessity of gathering all the data in flight on a small aircraft, and the second was the desire to use the existing wind tunnel technology and experience developed by the AARL staff.

The aircraft serving as the test bed for the flight program was a Beech C-23 Sundowner shown in Figure 1. When equipped with a GA(W)-2 "glove" airfoil over the existing wing structure the weight and stability requirements of this modified aircraft placed limitations on the weight and location of the data package. Further, the compact cabin restricted the size of the system and required a control panel to be designed for use by the flight test engineer to operate the instrumentation systems from the right front seat. Additional factors such as the available electrical power, exposure to temperature changes, the pressure fluctuations and the vibrations in the flight environment also influenced the design.

To obtain pressure distributions on airfoils, the blow-down transonic wind tunnel at AARL employs a scanning valve-transducer combination coupled to cut-off valves which trap the pressures input from the wind tunnel during a test run. As the tunnel test is terminated, usually after less than twenty seconds, the scanning valve steps through 48 ports, processing the transducer signals through a digital computer for near on-line data reduction. This data is used for lift and moment evaluations of airfoils while a traversing total pressure probe is used to measure momentum defect in the wake of airfoils and thus obtain airfoil drag. A point to note is that experience at AARL has shown that a single probe producing a continuous trace of the character of the wake provides more detail and accuracy than a multiple probe wake rake.

Use of an airborne acquisition package patterned after this AARL wind tunnel system provided several advantages. By trapping airfoil surface pressures at one instant, any subsequent flight disturbance did not influence the airfoil pressures; further, there would be no pressure response problem since adequate time on test condition could be assured before cut-off valve operation, allowing the scanning valve to step through the ports at a leisurely rate of one port per second after cut-off actuation. Although a wake rake was considered and would be compatible with the scanning valve-cut off package in use for the surface pressures, the advantages of the continuous trace and the results of an airfoil drag study reported by Saltzman (2) weighed in favor of the single traversing probe. Further, a multiple probe wake rake could not be

flown at the same time as an airfoil surface pressure flight since almost all of the scanning valve ports would be committed to the surface pressure data.

An additional feature of the wind tunnel type package was that the existing computer software would be available to guide the data reduction of the airfoil flight tests (3). Once software was developed for the analog-to-digital conversion of data from the FM tape recorder the entire data reduction process could be accomplished with the AARL Harris/5 digital computer system.

Other factors that were considered during the initial design phase included the availability of a seven channel FM tape recorder which was selected as the prime data recording instrument and the necessity to maintain an economical approach by using as much apparatus on hand at the AARL as possible. Fortunately, bridge balance and span control units and several pressure transducers were available to minimize instrumentation costs.

Integrating these initial concepts into the acquisition package led to the system shown schematically in Figure 2. Multiple surface pressures of the airfoil trapped by cut-off valves were measured by the scanning valve transducer and recorded on FM tape. Simultaneously, wake total and static pressures were measured and recorded from a survey probe rotated through the wake. Aircraft angle of attack and dynamic pressure were also obtained at this time. A more detailed description of the system and its components will now be presented.

#### AIRFOIL SURFACE PRESSURE INSTRUMENTATION

The instrumentation to obtain airfoil surface pressures consisted of five main components: the sensor(s), the cutoff-scanning system, a pressure transducer, a bridge balance and span control, and an amplifier.

The sensor to be used to obtain surface pressures posed the greatest problem. Since data had to be taken at several spanwise stations and each station required forty taps, flush taps were too expensive. The alternative was to use a belt of tubes wrapped chordwise about the airfoil; this method offered ease of fabrication and flexibility in location. The pressure belt was fashioned from two strips of ten-tube belt with a maximum thickness of .2 inches and a width of 4 inches. To preserve two-dimensionality near the belt, narrower dummy belts were placed on either side of the active belt bringing the total width to 5.5 inches. The twenty tubes forming the active belt would normally provide for twenty surface pressures. However, the proper positioning of a plug in each tube provided for forty surface pressures. The belt thus fabricated was attached to the wing of the test aircraft with tape gummed on both sides (ordinary carpet tape). The carpet tape method proved effective over the speed-pressure coefficient range of the test aircraft,

that is, for dynamic pressure less than .30 psi and pressure coefficients greater than -5.0. Figure 3 shows the pressure belt installed at the most outboard wing station.

Recognizing the potential influence to the local flow of this fabrication, the coordinates of the airfoil plus belt and the coordinates of the airfoil alone were input into a theoretical computation (4) to examine the effect of the pressure belt on the pressure coefficients. The theory indicated little difference in pressure coefficient and no appreciable change in lift coefficient or moment coefficient.

Completing the sensor fabrication was a "cable" lead-in pressure lines. The lead-in lines were 18 feet length strip-a-tube lines connecting the pressure belt to the cutoff scanning system. The cutoff scanning system, an off the shelf system having 48 inputs available, was able to accept the forty inputs from the pressure belt as well as 2 inputs for the measurement of dynamic pressure. This arrangement still left 6 ports (inputs) available for other pressures, as desired.

A DC-type pressure transducer was an integral part of the scanning valve. This single 2.5 psid transducer gave an analog output corresponding to each of the sensed pressures. Excitation and balancing for the transducer was provided by a wheatstone bridge arrangement in the bridge balance and span control unit, and the low level output was amplified by a custom built amplifier. Figure 4 shows the final configuration of the bridge balance and span control unit and the cutoff-scanning system.

#### AIRFOIL WAKE INSTRUMENTATION

The instrumentation to obtain airfoil wake pressures consisted of four main components: the total and static pressure probes, the driving mechanism and potentiometer, a total pressure transducer and a static pressure transducer. The pressure probes were fabricated such that both total and static pressures could be obtained at the same chordwise station. This meant each sensor was at a different spanwise location. Because the difference between static and total probes was only 2 inches, changes in spanwise direction were ignored.

The probes, shown in Figure 5, were fashioned from 1/8 inch stainless steel tubing and were reinforced with 3/8 inch tubing. The static probe was provided with four, .016 inch holes evenly distributed around its perimeter and the total probe was chamfered inside its open end. The leads from the sensors were run through and soldered to a 1/2 inch stainless tube, 44 inches long. This tube served as a drive shaft for the survey probe.

For simplicity in fabrication and mounting, the mode of motion of the probe was rotation, not translation. A small 28 VDC motor

was used to drive the probes, and a worm-driven gear arrangement was chosen to provide a high gearing ratio. A ten turn precision potentiometer was driven from the motor shaft to give an angular position through the wake. The entire range of the potentiometer was used to provide good resolution.

The drive mechanism and position potentiometer were mounted in an aluminum box; the drive shaft was inserted through the end and mated to the drive mechanism. The small bearing shown in Figure 5 was mounted from the trailing edge and used to prevent potential shaft vibration; the box was streamlined with foam formers to prevent gross interference.

To sense the wake pressures and provide good time response, the total and static pressure transducers were mounted in the wing of the aircraft in close proximity to the probing system. AC type transducers and modified carrier demodulator units were used since they were already available. The static pressure was measured with a 2.0 psid transducer while a 1.0 psid transducer was used to measure total pressure. The reference for both transducers was the aircraft total pressure system.

#### AIRCRAFT ATTITUDE INSTRUMENTATION

To reduce and correlate the data it was necessary to have aircraft speed and angle of attack measurements. The sensor used to provide test aircraft speed (dynamic pressure) was a fully swiveling pitot probe. The probe was mounted on a boom extended approximately 3 feet from the leading edge of the left wing of the test aircraft. Due to its freedom of motion and vane configuration the probe was always aligned to the relative wind thereby eliminating errors due to probe angularity. The probe static pressure lead-in was connected to the cutoff-scanning system at two ports giving redundancy in this crucial measurement. The probe total pressure was used as a reference pressure.

The angle of attack sensor shown in Figure 6 consisted of a vane 6 inches long fitted with a small fin. The vane was connected to a one turn potentiometer via a 1/8 inch shaft. The device was mounted to a boom extending from the right wing of the test aircraft. Vane output was correlated with an inclinometer mounted in the cabin of the test aircraft. The inclinometer consisted of an aluminum arm 10 inches in length, fitted with a bubble level. Mounted on a graduated backplane, the level arm-bubble level combination provided aircraft attitude to  $\pm 0.2$  degrees.

#### RECORDING INSTRUMENTATION

A Lockheed Model 417 FM Tape Recorder was used to save all types of data. This system offered 7 data recording channels and a voice track with a total weight of 29 pounds. Each data channel had a range of  $\pm 1.4$  volts and frequency response in excess of 15 KHz.



The recorder was powered by a 16.8 V rechargeable battery pack and was fastened to the top of the instrument pack by Bungee Cord, for easy installation and removal.

#### CONTROLLING AND POWER EQUIPMENT

The most important item used for the proper operation of the instrumentation was the remote controller shown in Figure 7. Contained in the unit were 4 main circuits:

- i the cutoff-scanning system stepping circuit,
- ii the wake probe circuit,
- iii the amplifier circuit, and
- iv the data identifier circuit.

The cutoff-scanning system stepping circuit consisted of an integrated circuit timing chip driving a 24 volt reed relay and a digital counter. The period of the timing pulse and the pulse width were variable. The timing circuit was used to automatically step the scanning valve through the number of ports specified by the counter. Provisions were also made for manual stepping of the scanning valve. The cutoff valves were opened and closed only manually.

The wake probe circuit consisted of a rheostat and an UP-OFF-DOWN switch. This circuit provided manual operation of the wake probe with both direction and speed control. Voltage output to the drive motor could be varied from 0 to  $\pm 28$  VDC, thereby allowing adequate control of probe speed.

The amplifier circuit was composed of an integrated circuit amplifier and emitter follower. The emitter follower was used to allow for high source impedance and low output impedance. It was needed since the scanning valve transducer was a high (20 K $\Omega$ ) source impedance device. The amplifier had variable gain and offset correction potentiometers giving a maximum gain of 80 and excellent zeroing capability.

The data identifier circuits were used to aid in data reduction. They were simply relays which gave a "high" when the cutoff valves were closed and each time the scanning valve stepped. This gave a method for determining valid data and scanning valve port number.

The electrical power required to operate the instruments was obtained from DC-DC converters. The aircraft system was a nominal 14 VDC and the converters were used to give 5 VDC, 12 VDC, 24 VDC, 28 VDC and  $\pm 15$  VDC. Voltage regulators were used to hold these voltages to better than  $\pm 1\%$ .



## CALIBRATION OF INSTRUMENTS

All data voltages had to be scaled to  $\pm 1$  VDC since ground based reduction hardware (A/D converter) was limited to  $\pm 1$  VDC. Three pressure transducers and two potentiometers had to be calibrated to insure this proper output voltage range.

The scanning valve transducer was calibrated by simply applying the maximum gain of the amplifier. The transducer was a 2.5 psid and expected maximum pressures were on the order of .5 psid, i.e., 20% of the possible dynamic range. With a gain of 80 applied, the slope became .88 psi/volt. The resulting zero output viewed on an oscilloscope revealed a .005 volt uncertainty. Based on a full scale voltage of 1.0 volt, this uncertainty corresponded to  $\pm 0.5\%$ , an acceptable percentage error.

The two wake probe transducers were AC type. Excitation, span control and zeroing were supplied via carrier demodulator units. The 2 psid transducer was calibrated to .50 psi/volt and was used in the wake static pressure system. The 1.0 psid transducer was calibrated to .25 psi/volt and was used in the wake total pressure system. Each transducer provided outputs to  $\pm .2\%$  of full scale and had aircraft total pressure as reference.

To calibrate the angle of attack potentiometer, the sensor vane was first fitted with a collar which limited its total travel to 40 degrees. A mark was then made to indicate a zero position, this configuration made calibration a simple matter of taking three voltage readings, one corresponding to each of the three positions. The calibration slope and zero could then be determined. For the excitation voltage applied (5 volts) the resulting calibration slope was 64 degrees/volt.

The wake probe potentiometer was calibrated in a similar manner with the calibration slope being 220 deg/volt. This large slope was the result of the fact that the probe could survey the wake at two spanwise stations, 180 degrees from each other. (As will be shown later, this situation caused severe problems).

The FM tape recorder was another instrument to be calibrated. All channels were checked for linearity and bias. It was found that shorted inputs resulted in a time average zero output however, there was a + 20 millivolt oscillation in output. This situation was acceptable only if a long enough time average could be computed. A sufficient time average could be obtained with scanning valve data since the time spent recording each port was adjustable. However it was foreseen that problems with the wake survey measurements could develop due to the changes inherent in that type of data; that is, it was possible that a useable time average did not exist for the wake surveys. Actual flights were needed to confirm the fact (it indeed became necessary to change the method of probing).

One non-electrical calibration had to be made to obtain the mass carry over correction factor for the cutoff-scanning system. Using an isothermal assumption, an equation to represent the correction for the trapped volume may be written:

$$P_1 = p_1(1+C) - p_{1-1}(C)$$

where:  $P_1$  = actual trapped pressure at 1<sup>th</sup> port

$p_1$  = sensed pressure at 1<sup>th</sup> port

$C = \frac{\text{transducer volume}}{\text{trapped volume}} = \text{constant since both volumes were constant for all } i$

The factor was found by trapping a pressure in one port while trapping other pressures in the next two ports; the pressures in the last two being equal to each other but different from the first. By scanning through the ports and recording the respective voltage outputs, the sensed pressure values were obtained. Since the scanning valve was already open to the first port, no correction was involved with it. However, since some mass was carried into the subsequent ports through the scanning procedure, and it was known that those ports initially had equal pressures, 2 equations could be written, set equal, and solved for C. This procedure resulted in a value of .06 for the mass carry over correction factor.

The last calibration was a correlation between inclinometer reading and the output of the angle of attack potentiometer. The correlation was determined by flight test. The result was a linear curve relating aircraft reference angle and local angle of attack and having a slope of .685 with a zero that varied depending on the particular wing station under consideration.

#### TYPICAL FLIGHT USE

To bring the complete acquisition system into better focus, consider a typical test flight and the operations necessary to acquire the surface pressure and wake data.

The rear seats of the test aircraft were removed to provide room for the data acquisition package. The package, with overall dimensions of 16" x 19" and 21" high including the recorder, was bolted to vibration mounts on the cabin floor and centered directly behind the flight crew. Weight of this package, again including the flight recorder, was 79 lbs.

Prior to each test flight a check of the operation of all the instrumentation was made, with the survey probe, cut-off and scanning valve cycled through their operating ranges from the remote controller. The automatic sequence control for the tape recorder and the scanning valve was examined and all instrumentation calibrated. These calibrations included recording the voltage outputs

from the angle of attack vane at its upper and lower stop and aircraft level positions, the outputs from the wake survey probe at its lower and upper position and at a wing aligned reference position, and the output of all transducers.

After takeoff and during the 5 to 10 minute flight to the test area, a pre-run calibration was made to record the transducer zeros and the output of calibrate resistors built into the three transducer circuits that were used to impress the calibration of the transducers on the data tape.

To obtain the zeros, the pressure on both sides of the transducers must be equal. This was accomplished easily with the scanning-valve transducer by connecting a balance line across one port of the cut-off valves. When the cut-off was closed, reference pressure - the aircraft total pressure - was applied to both sides of the scanning valve transducer. With the wake probe, however, the balancing process was not as simple, since the probe total and static orifices were exposed to the airstream. A guillotine type valve was built and installed in the wing, next to the two wake pressure transducers. In normal operation, the guillotine closed a balance line from the transducer reference to the measuring side. During the flight calibration a solenoid opened this balance line and closed the input from the probe ports, allowing the pressures to equilibrate on both sides of the transducer diaphragm and the zero output to be recorded. This guillotine system proved troublesome and was eventually replaced and the zeroing technique modified. These details will be discussed subsequently.

With this pre-run calibration complete, and the aircraft at test altitude, usually 5000 ft, the aircraft was trimmed level at a specified airspeed - from 120 mph to 60 mph. When the aircraft settled out, the cut-off valves were closed on the pilot's "mark" and the automatic sequence initiated. This sequence started the recorder, delayed 15 seconds to allow the recorder to come up to speed, then triggered the scan sequence. While the scanning valve stepped through the 48 ports in 48 seconds, the test engineer operated the wake survey probe, traversing a six inch region behind the wing in approximately six seconds. Four passes of the wake survey probe were usually obtained during each run. After the scanning valve completed its cycle, the recorder was turned off automatically and the cut-off valves opened to make ready for a new test point. The pilot proceeded to the next airspeed and the process was repeated. Each test point required between three and four minutes to establish a trim, level flight condition and to cycle the instrumentation, and a typical test flight produced 12 to 13 test points. The complete flight, which included a second flight calibration prior to the final test airspeed, took a little more than one hour flight time.

Once on the ground, the tape recorder was removed from the aircraft, carried to the Aeronautical and Astronautical Research Laboratory and the data processed.

## DATA REDUCTION

Relative simplicity in data reduction was necessary for practical application of the data acquisition system. Since large quantities of data were developed from each flight - more than 400 surface pressures and as many as 40 wake surveys - efficient processing was required. The key to efficient handling of the data was the two data identification traces supplied by the controller. The Harris/5 computer was used for data reduction and these two traces, the cut-off "high" and the scanning valve port identifier, provided the necessary interface between the analog FM recorder and the computer's 16 channel analog-to-digital converter. Using this method a set of 17 data runs could be processed from raw form on analog tape to hard copy plots of pressure coefficient and lift, drag, and moment coefficient in less than 3 hours.

As indicated in Figure 8, the data handling proceeded as follows. A program was coded which digitized the analog tape and stored the data in discrete form on a digital magnetic tape. The cutoff identifier trace was used to indicate where to start-stop the digitizing. Each run was identified by a header and calibration data was identified by a header different from run data. This phase of reduction required one of the three total hours. After this digitizing process, reduction was independent of the analog recording system.

The next program in the sequence was used to sort the digitized data and put it in a useable form. The scanning valve port identifier came into play here. Once isolated, the identifier data was searched for the "highs" indicating a step to the next port. Since the width of each data window was fixed, it was possible to then average over the window and obtain the raw voltages for each sensed pressure of the scanning valve. At the same time, the total and static pressure voltages and position voltages through the wake were identified and isolated by an appropriate algorithm; this procedure was automated, but monitored on the CRT display. If the data was acceptable, it was written to a disc file and made available to the final program in the reduction sequence.

The last program converted voltages to pressures and position and converted all data into normally accepted forms, for example, airfoil surface pressure in the form of pressure coefficient, as shown in Figure 9, or wake pressures as indicated in Figure 10. The lift and moment coefficients were integrated from this surface pressure data and the wake survey pressures were integrated to obtain drag coefficient. Hard copy plots and printouts were provided for easy analysis of the data. This reduction phase in conjunction with the data sorting required the other 2 hours of the total 3 hour time.

## OPERATIONAL EXPERIENCE

To date, the acquisition package has been flown about twenty-five hours on the test aircraft, completing the first phase of the GA(W)-2 evaluation. During this time there has been remarkably little trouble with the equipment, the controller, and the cut-off and scanvalve operating as designed. A few operational problems have been encountered, however, which are of interest and warrant comment.

After the first flight, for example, spurious signals appeared on the data tape at apparently random intervals. A second flight identified the source as radio transmissions from the aircraft; therefore, every subsequent flight was made with no transmissions during the data acquisition process.

The only problem created by the cutoff-scanning system was pressure leakage once the cutoff valves were closed. The system was checked for leaks with the cutoff-valves closed prior to installation and leaks, if any, in terms of voltage were on the order of .1 millivolt per second. However, initial run data reduction revealed 4 leaking ports, judged so by their obvious deviation from a smooth trend in the airfoil surface pressure data. The problem was traced to a loose face plate on the scanning valve and the situation was easily corrected by the tightening of 3 bolts. As a consequence, the tightness of the scanning valve face plate became an item on the pre-flight checklist.

Another problem was identified when the wake probe was activated. When the  $\pm 20$  millivolt flutter of the recorder was superimposed on the probe position signal, the position of the probe could not be determined within a  $\pm 1/2$  inch. This was a result of the initial design concept of using the probe to traverse through more than  $200^\circ$  of arc so that two spanwise stations could be surveyed during one flight. By restricting the probe arc to  $40^\circ$  and by introducing a bias voltage of -1 volt to take advantage of the recorder  $\pm 1$  volt capability, position error was reduced to better than  $\pm 0.1$  inch.

Wake static pressure, which is close to free stream static, was difficult to measure. Originally arranged with the aircraft total pressure as the reference pressure for the static pressure transducer, the reference pressure for the total and static pressure transducers was changed after a few flights, to the aircraft static pressure. The first static pressure transducer was then replaced with a more sensitive 0.1 psid transducer to record the minor variations of static pressure (less than 0.01 psi) through the wake.

When it became necessary to fly the aircraft during the winter months, temperatures below  $20^\circ\text{F}$  were imposed on the wing installed pressure transducers. Calibrations of the transducer made at depressed temperatures before installation showed slope changes near 4% at this temperature. The transducer zero also shifted several millivolts with decreasing temperature; when coupled with the

unreliable guillotine the transducer zero was difficult to ascertain, and the temperature effects jeopardized the accuracy of the wake probe measurements. To resolve this problem, it was necessary to move the transducers, temporarily, from the wing to the protected environment of the cabin.

While this transducer location eased the temperature problem and at the same time made manual valving possible to balance the pressures for ready transducer zeroes, the long pressure leads from the wing mounted probe to the cabin introduced a potential pressure response problem. An experiment verified the problem, showing the pressure lines responded to a step pressure input with a time constant of 0.1 seconds. It became necessary, therefore, to modify the continuous probing and to resort to a rotate-and-pause intermittent action of the wake probe. While not ideal, the approach did restore the reliability of static pressure measurements. A return to the continuous probing method of wake evaluation is anticipated as temperatures warm and a more reliable guillotine system is employed.

#### SUMMARY

The design of a data acquisition system in use for the flight evaluation of the GA(W)-2 has been presented. Novel design features of this portable system are the cutoff-scanning valve arrangement, the automated remote control and the efficient interface between the analog tape recorder and the digital computer used for rapid data reduction. Coupled with the pressure belt taped to the wing of the test aircraft the system has been used to measure surface pressure distributions with excellent results. Simultaneous probing of the airfoil wake yields drag data of similar quality.

Operation of the system has been relatively free of problems and all those worthy of note have been considered. In summary, then, the acquisition system has performed as designed in an economical, timely and cost-effective manner.

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FIGURE 1. THE BEECH SUNDOWNER - TEST BED  
FOR THE GA(W)-2

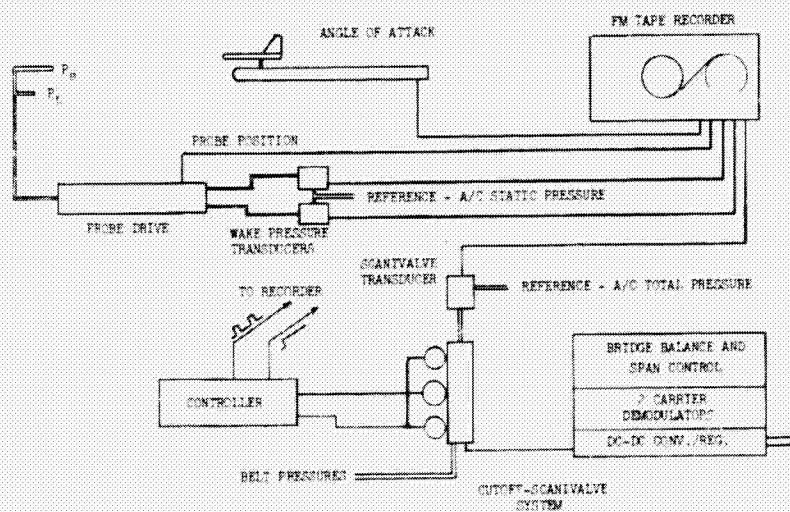


FIGURE 2. SCHEMATIC OF THE DATA ACQUISITION SYSTEM



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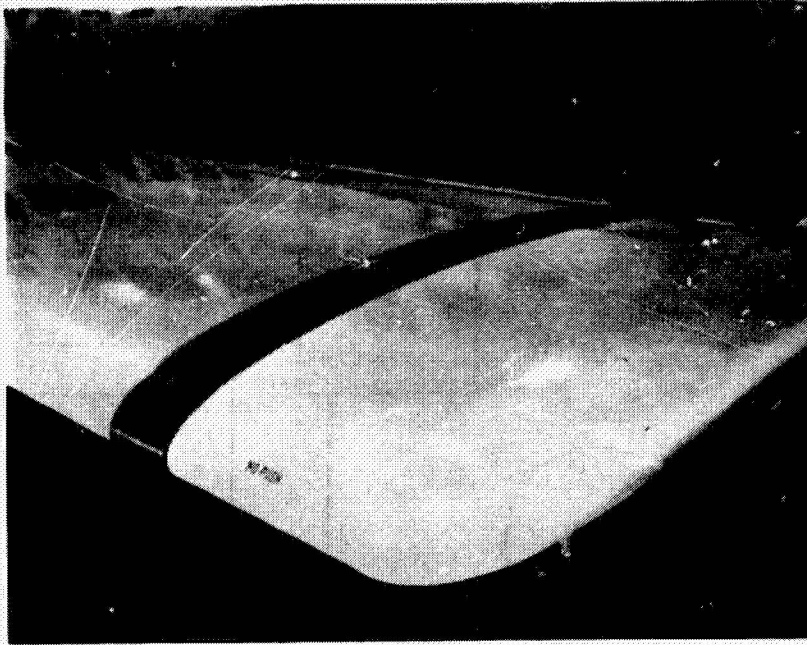


FIGURE 3. PRESSURE BELT INSTALLED ON WING

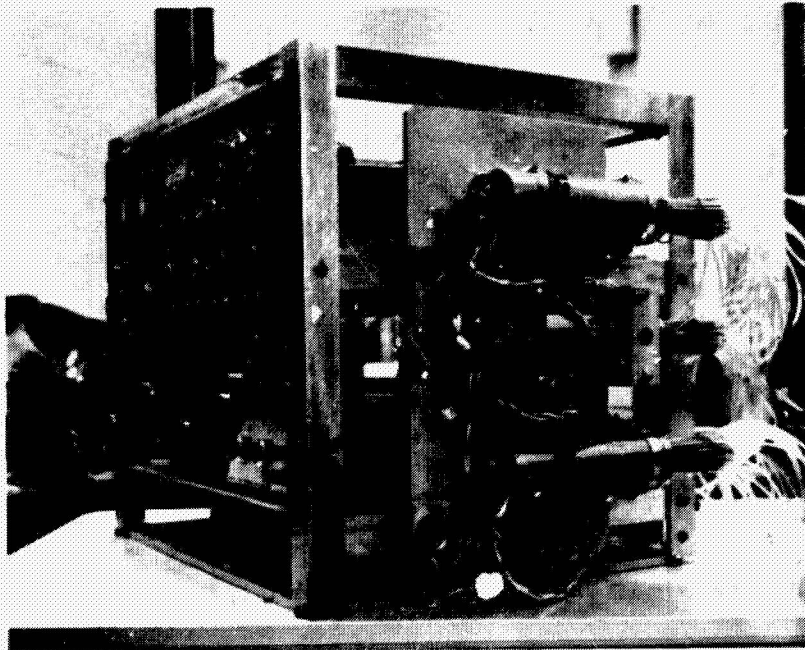


FIGURE 4. THE INSTRUMENT PACKAGE



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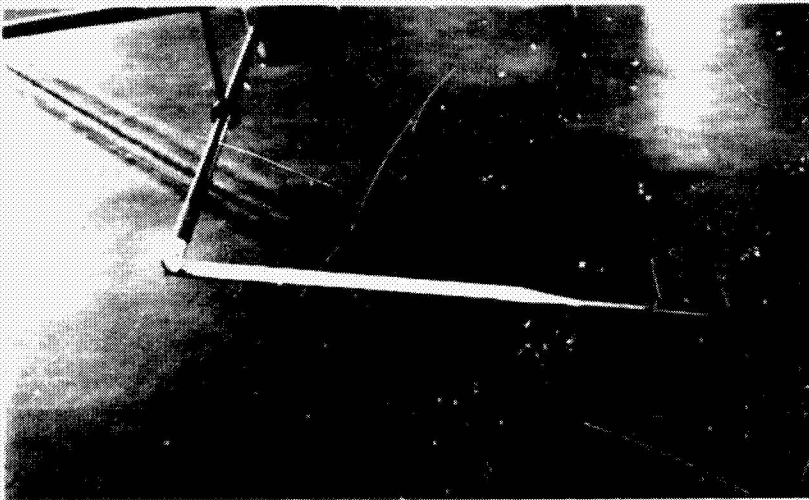


FIGURE 5.

WAKE SURVEY PROBE



FIGURE 6.

ANGLE OF ATTACK SENSOR

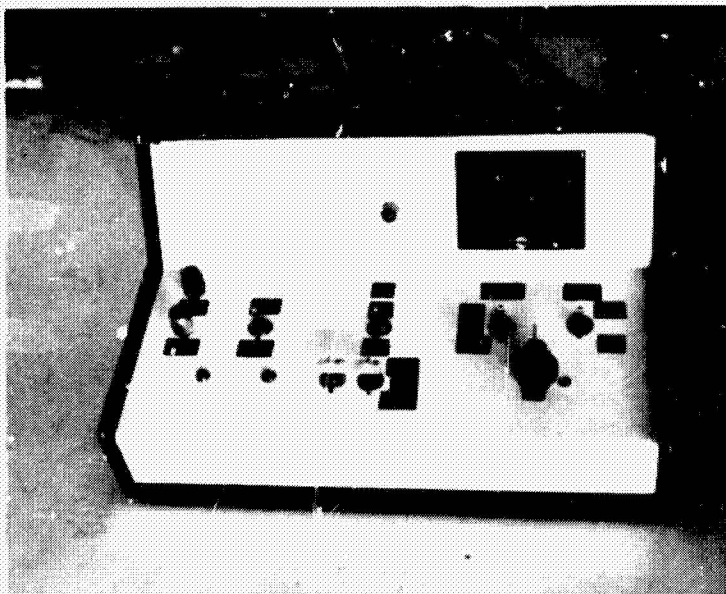


FIGURE 7.

TEST ENGINEER'S  
REMOTE CONTROL PANEL

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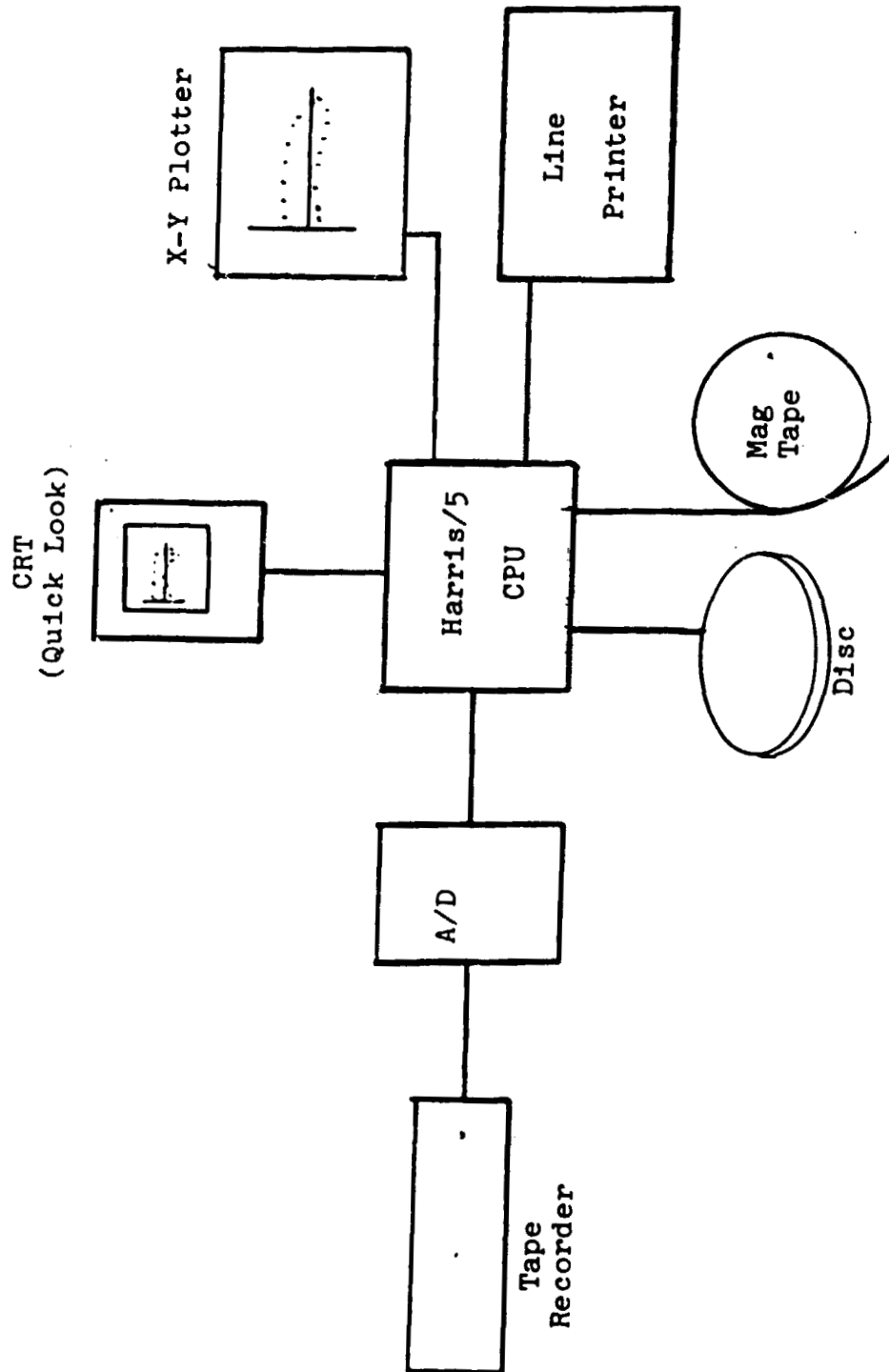


FIGURE 8. SCHEMATIC OF DATA REDUCTION

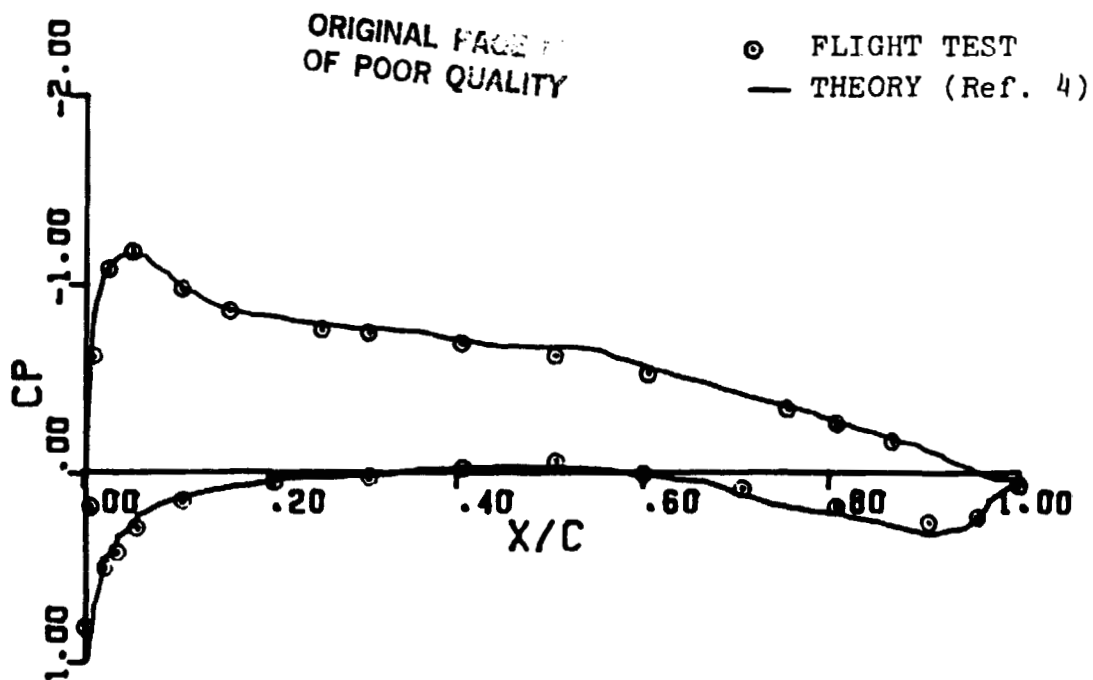


FIGURE 9. COMPARISON OF AN EXPERIMENTAL PRESSURE DISTRIBUTION WITH THEORY

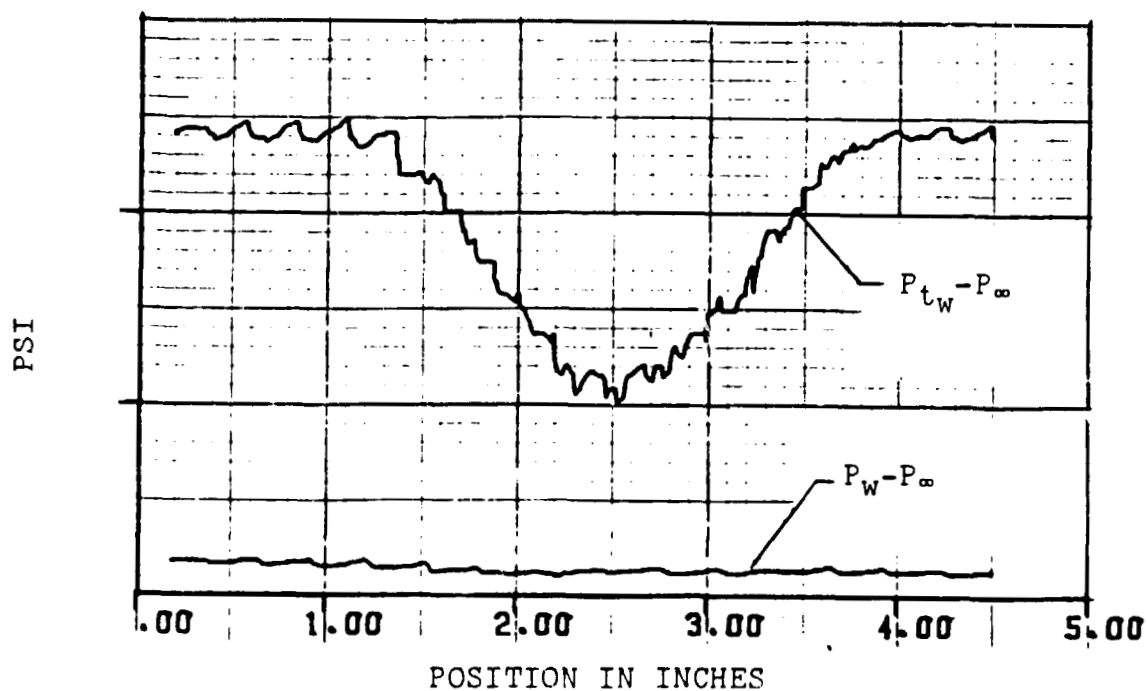


FIGURE 10. TYPICAL AIRFOIL WAKE SURVEY

FLIGHT TEST TECHNIQUES  
FOR LOW SPEED AIRFOIL EVALUATION\*

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SUMMARY

Techniques for in-flight evaluation of new airfoils by modifying a single engine general aviation aircraft and measuring and recording airfoil surface pressures, airfoil wake pressures, and aircraft angle of attack and airspeed are presented. Included are descriptions of the aircraft modifications, instrumentation, data reduction techniques, illustrations of typical results and comments on new equipment for flight test applications.

INTRODUCTION

The NASA LS(1)-0413 airfoil section characteristics have been evaluated in a flight test program (ref. 1). A single engine aircraft was modified, instrumented and flown, and pressure data was acquired, reduced and summarized. The program was successfully accomplished by implementing certain effective techniques. The existing wing of a "Beech Sundowner" testbed was "gloved" over the existing full-span to the contour of the LS(1)-0413 airfoil by the Beech Aircraft Corporation, an active participant in the flight program, thereby saving the expense of construction of an entirely new wing. The aircraft was instrumented with existing equipment supplemented by specially developed pressure measuring systems. A sophisticated and efficient data processing scheme was developed to handle the large quantities of data.

The purpose of this paper is to summarize the details of the techniques used to effectively complete the flight test program and to comment on some new instrumentation systems that could enhance future flight test efforts.

The symbols used herein are defined in an appendix.

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\*Some of the techniques described herein were developed and implemented while under contract to NASA Langley Research Center.

## AIRCRAFT MODIFICATION

### Background

To satisfy the objective of determining the section characteristics of the LS(1)-0413 airfoil in flight, it was first necessary to change the wing of a testbed aircraft, shown in figure 1, to the appropriate contour. Two options were available; build an entirely new wing, or modify the existing wing. Designing, building and proving a new wing is, unfortunately, time consuming and expensive. Modifying the existing wing by bonding on a new surface seemed attractive for this particular program for various reasons:

- 1) budget constraints favored the relatively inexpensive nature of this modification approach,
- 2) structural integrity of the existing wing structure could be utilized,
- 3) cable and tubing routing could be easily facilitated,
- 4) total time to modify the aircraft wing was a fraction of the time of the former alternative,
- 5) by use of bonding, a smooth finished surface could be obtained without extensive structural proving.

Based on these observations the "gloving" approach to wing modification was adopted. This simply meant that the new contour would be obtained by bonding formers to the old wing surface and bonding a new skin to those formers.

### Design and Fabrication

The LS(1)-0413 was a 13% thickness ratio airfoil while the existing 632415 airfoil of the Sundowner was 15%. Figure 2 shows the modified wing with the larger chord. In order to accommodate the modification in the easiest manner, the leading edge was extended by 17.5 cm (7 in.) and the trailing edge by 7.5 cm (3 in.) to obtain a "good" range of center of pressure relative to the aircraft center of gravity (CG). The new gloved wing had an incidence angle 1.4 degrees larger than the existing wing, and the original linear wing twist of 2 degrees washout from root to tip was preserved. Also, an internal channel was incorporated into the modification for running cables and tubing from the wing without significantly disturbing the airflow.

The modification was begun by stripping the painted surfaces of the existing wing and ailerons. Balsa formers of 2.5 cm (1 in.) thickness were then bonded to the "old" skin on 20.3 cm (8 in.) centers with an epoxy type adhesive. Spanwise stringers were used for contour uniformity with wing results as shown

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in figure 3 and aileron results in figure 4. These formers were wrapped with 0.5 mm (0.020 in.) sheet aluminum which was also bonded, thereby leaving a smooth, uniform rivetless surface. Also, external mass balances were used to statically balance the now modified ailerons.

## INSTRUMENTATION SYSTEM

### Overview

Once modified, the aircraft was instrumented as shown schematically in figure 5. Aircraft angle of attack was monitored as was dynamic pressure (not pictured). A scanivalve/cut-off valve system, operated by a remote controller, was used to acquire surface pressure data. A wake survey probe, sensing total and static pressures, and a rotary drive mechanism were used to obtain momentum deficit information, ultimately resulting in drag coefficients. All these systems were powered from an instrument rack equipped with power supplies and signal conditioning equipment.

### Sensors

The angle of attack sensor (vane) pictured in figure 6 consisted of a 15.2 cm (6 in.) stem fitted with a 3.8 cm (1.5 in.) fin, driving a one turn potentiometer. A collar was used to limit the sensor travel to 5 degrees nose up and 40 degrees nose down relative to the mounting boom. The vane was located 0.75 chord ahead of the quarter chord point. To correlate the vane reading to local section, a deck angle inclinometer (bubble level, figure 7) was used. This added piece of instrumentation allowed for the determination of local geometric angle of attack.

To sense surface pressures along the chord a strip-a-tube belt was used. The belt was formed from 5.1 mm (0.2 in.) plastic tube arranged in a group of twenty. Each tube was plugged appropriately to obtain forty active lines from the twenty tubes. Orifices were located once the belt was fastened to the aircraft wing surface with double sided tape. Lead in lines were run from the belt sensor, through the wing channel, into the cabin and connected to the scanivalve/cut-off valve system. Figure 8 shows a typical belt installation.

The wake survey probe was actually two sensors - one total pressure and one static pressure sensor - which was rotated through the wake of the wing at one of two spanwise stations during a given flight. The sensors were separated from each other by 5.1 cm (2 in.) in the wing spanwise direction, thus allowing total and static pressures to be measured at the same chordwise station (0.14 chord aft of trailing edge). By properly locating the wake probe, either a baseline (basic airfoil) or aileron station could be surveyed. Figure 9 illustrates this wake survey probe.

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Supporting Equipment

To provide excitation voltages and signal conditioning an instrument rack (figure 10) was constructed. The rack contained all necessary power supplies, amplifiers, bridge balance units and carrier demodulators. Also, the scanivalve/cut-off valve system was mounted to the rack, and the 7 channel FM analog tape recorder used for data recording was mounted atop the rack. The total weight of the rack fully equipped was 351 N (79 lbs.) with approximate dimensions of 48 cm W x 53 cm H x 41 cm D (19 in. W x 21 in. H x 16 in. D).

For ease in operating all the instrumentation systems a remote controller was designed and built. The controller allowed full manual or automatic control of the systems from the flight test engineer position. The controller had an internal clock used to sequence the scanivalve/cut-off system and the "pitch-pause" motion (described later) of the wake survey probe. Once the engineer selected a data point, he did not have to intervene until all data from that test point was fully acquired and recorded. The systems would then be reset for another test condition. Figure 11 is a photograph of this controller.

DATA PROCESSING

A typical test flight would produce twelve test points resulting in large quantities of raw data. A special data processing scheme was developed to efficiently handle these data. Upon completion of a flight, the FM recorder (figure 12) would be taken to the ground based digital computer system (figure 13) and would be "patched" into the computer, thereby allowing the computer to digitize the analog signals played back by the recorder. Timing pulses provided by the instrumentation controller greatly assisted in the digitizing sequence. FORTRAN coded programs were used to manipulate the now digitized data and also allowed the operator to select options as to how the data should be reduced and presented. For instance, plots and printouts could be immediately generated for each test condition and/or summary plots could be made. In typical cases, fully reduced and plotted data could be "in-hand" within 3 hours of aircraft landing.

TYPICAL RESULTS

Surface Pressures

Typical of the partially reduced surface pressure data were pressure coefficient-chordwise location plots shown in figure 14. Two flight test angles of attack are shown by symbols and the corresponding analytic computations (ref. 2) are shown by the solid lines. These comparisons are at matched angle of attack, not matched lift. The two-dimensional angle of attack was obtained by subtracting an induced angle calculated by a three dimensional analytic code of Beech Aircraft Corporation. At the lower angle the scatter in the flight test data is seen to be low and compares well with analytic calculations. At

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the higher angle (lower dynamic pressure for the case at hand) the scatter is slightly more due to the pressure belt itself and the decreasing accuracy of the pressure measurements at low dynamic pressure. Overall, however, these results are highly acceptable.

### Wake Surveys

Partial reduction of the raw data from wake surveys lead to pressure-position plots similar to those shown in figure 15. The first plot shows a continuously scanned survey while the second plot shows the stepped or "pitch-pause" method of surveying. The latter method was generally used to eliminate any potential response problems. The result of the method was a physical averaging of the data due to a finite number of points through the wake (shown by the relative smoothness of the plot). Both total and static pressures were measured and presented and the static pressure variation is seen as significantly different from free stream static (the reference pressure). Based on these kinds of plots, limits of integration were chosen and drag coefficients produced.

### Baseline Lift and Moment

Carrying the surface pressure data reduction to completion by integration of pressure distribution resulted in lift and moment coefficients as functions of angle of attack (figure 16). Lift data from three spanwise stations are shown to coincide very well by applying the three-dimensional analytic induced angle correction and compares well with the faired wind tunnel data (solid line) of McGhee, et al. (ref. 3). The lift coefficient data becomes somewhat scattered at low dynamic pressures again due to the lower accuracy of the transducers in that regime. A small error in the dynamic pressure measurement unfortunately comes through strongly in the final reduction. The moment coefficients are scattered and are generally more positive than wind tunnel measurements. The significant deviation could be due to slight trailing edge differences between the LS(1)-0413 modified aircraft wing and the wind tunnel model used for comparison.

### Drag Polars

Full reduction of the wake pressures lead to the baseline drag polar shown in figure 17. The symbols represent two test flights taken almost one year apart. The wind tunnel data is again that of McGhee. The cases shown are for smooth wing surface and smooth model. Due to the varying Reynolds number in the flight data two bracketing wind tunnel cases are presented and the proper trend of the flight data can be seen. A similar drag polar is shown in figure 18. Here, however, the wind tunnel model boundary layer was tripped at 7.5% chord as was the flight test airfoil. The baseline drag polar is also shown for reference as a solid line. Very good agreement can be seen as the flight test data trends from almost exact agreement at lower lift/higher Reynolds number to good agreement at higher lift/lower Reynolds number.



## Aileron Station Data

Pressure belt data and wake surveys were also taken at aileron stations. Figure 19 shows a typical pressure distribution at an aileron station with the solid line being a fairing of the data for clarity. The "pinched in" (reduced lift) nature of the distribution due to gap flow is evident. No attempt was made to apply the three dimensional analytic induced angle computations at these stations due to this "different" type of flow phenomenon. Figure 20 shows the rather large increases in drag incurred by having gap flow. The comparison is with flight test basic airfoil (solid line) and represents a 45% increase in drag at cruise lift.

## NEW SYSTEMS

The additional experience gained by developing and using the aforementioned instrumentation system has led to the design and construction of at least two new useful instruments. Figure 21 shows a rotating type probe mechanism which is direct drive and adjustable in the chordwise direction. The rotating nature of the drive allows two spanwise stations to be surveyed. The drive itself is only 10.2 cm (4 in.) wide and high and 30.5 cm (12 in.) long. Overall length including the support platform is 55.9 cm (22 in.) while the total weight including a typical probe and foam formers (streamlining) is 26.7 N (6 lbs.).

To enhance data acquisition and reduction capability while in flight, a Digital Data Acquisition and Reduction System (DDARS) has been developed in-house (ref. 4). Figure 22 shows the main components of this system, the mainframe (on left), the operator's console, and the ground based disc drives (on right). The mainframe holds a 16 bit microprocessor, 32K words of memory, all necessary interfaces, dual cartridge tape drives and a 32 channel analog-to-digital conversion system (1 KHz effective throughput using all channels). Also included are ten program controllable relays for on/off type controls. Contained in a separate small rack (not shown) are 6 bridge balance units and 6 differential amplifiers for signal conditioning. The components used in a flight situation are the mainframe, the signal conditioner rack and the small console terminal. The total weight of the airborne package is 600 N (135 lbs.) and the maximum power consumption is 400 watts. The total volume required to locate the system is approximately 0.25 cubic meters (8.8 cubic ft.).

## CONCLUSIONS AND OBSERVATIONS

Effective techniques have been used to evaluate the performance of an airfoil in a flight environment. Aircraft modification and instrumentation have been discussed as have the data processing scheme and typical results. The data along with comparisons seem to indicate the system was acceptable over most of the aircraft speed range, becoming questionable only at the very low dynamic pressure end. Appropriate transducer replacement could cure that problem. Also discussed were two new systems which can be effectively applied

in future flight test efforts - a wake survey drive mechanism and a new digital data acquisition and reduction system. The DDARS system seems to offer a quantum improvement in flight test data acquisition and reduction by implementation of its microprocessor-based mainframe and real time peripherals while the wake survey mechanism offers good probe motion in a small, light package.

## REFERENCES

1. Gregorek, G. M., Hoffmann, M. J., Weislogel, G. S., and Vogel, G. M., "In-Flight Measurements of the GA(W)-2 Aerodynamic Characteristics", Paper 770461 presented at the SAE Business Aircraft Meeting, Wichita, Kansas, March 1977.
2. Stevens, W. H., Goradia, S. H., Branden, J. A., "Mathematical Model For Two-Dimensional Multi-Component Airfoils in Viscous Flow", NACA CR 1843, 1971.
3. McGhee, R. J., Beasley, W. D., Somers, D. M., "Low Speed Aerodynamic Characteristics of a 13 Percent Thick Airfoil Section Designed for General Aviation Applications", NASA TM X-72697, 1975.
4. Freuler, R. J., Hoffmann, M. J., "Digital Data Acquisition and Reduction System for Flight Testing General Aviation Aircraft", AIAA Paper No. 77-1216 presented at the AIAA Aircraft Systems and Technology Meeting, Seattle, Washington, August 1977.

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## APPENDIX

### SYMBOLS

Measurements and calculations were made in the U.S. Customary Units. They are presented herein in the International System of Units (SI) with the equivalent values given parenthetically in the U.S. Customary Units.

$c$	chord of an airfoil
$C_d$	section drag coefficient, $\frac{\text{Section drag}}{q_\infty c}$
$C_l$	section lift coefficient, $\frac{\text{Section lift}}{q_\infty c}$
$C_m$	section pitching moment coefficient with respect to 0.25 chord, $\frac{\text{Section moment}}{q_\infty c^2}$
$C_p$	static pressure coefficient, $\frac{p - p_\infty}{q_\infty}$
$p$	measured local pressure
$p_\infty$	free stream static pressure
$q_\infty$	free stream dynamic pressure
$R_e$	Reynolds number based on chord
$x, x/c$	distance along chord, non-dimensional distance along chord
$\alpha, \alpha_{2D}$	angle of attack in two-dimensional flow

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Figure 1.- Testbed aircraft in flight.

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Figure 2.- Airfoil-modification cross section.

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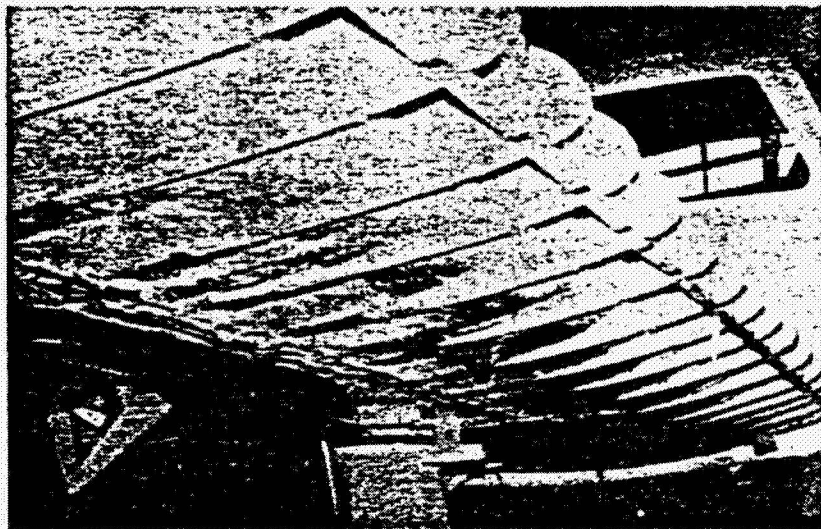


Figure 3.- Balsa-wood wing formers.

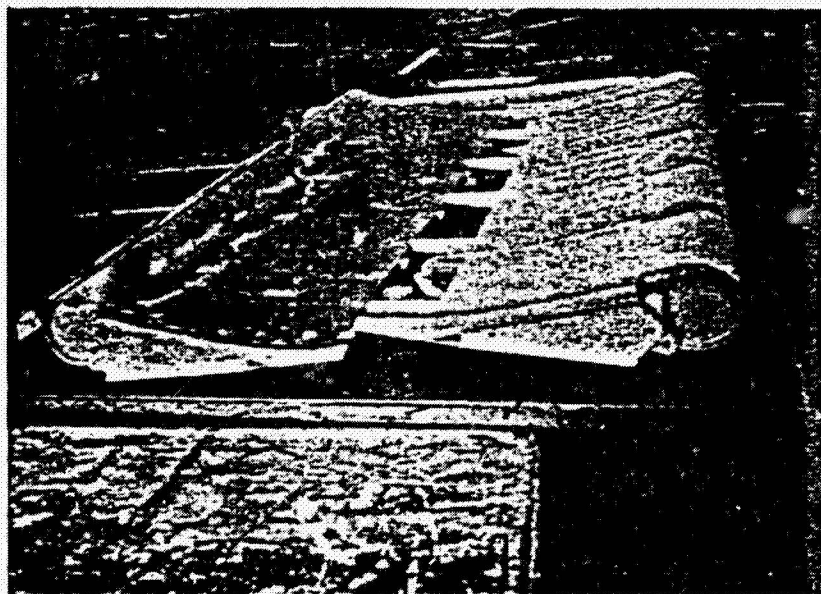


Figure 4.- Aileron modification.

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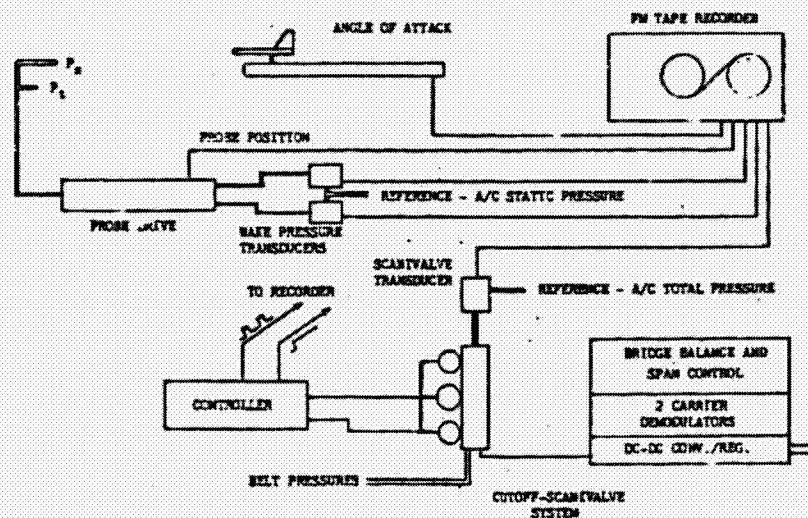


Figure 5.- Data-acquisition schematic diagram.

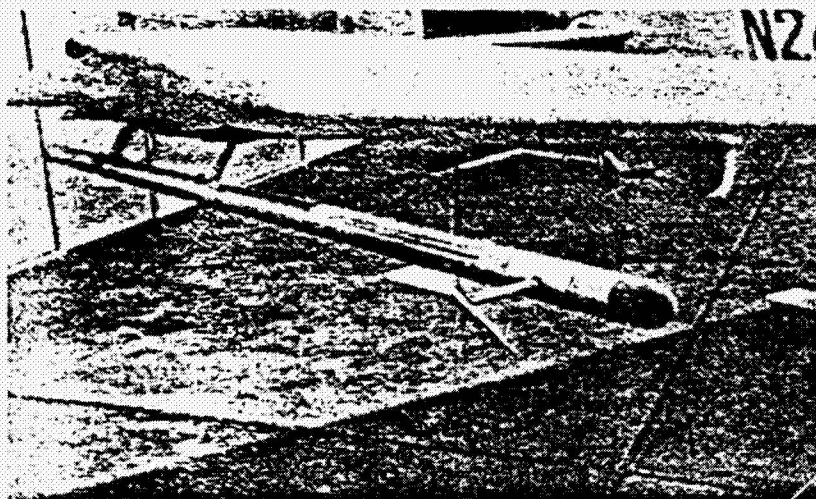


Figure 6.- Angle-of-attack sensor.



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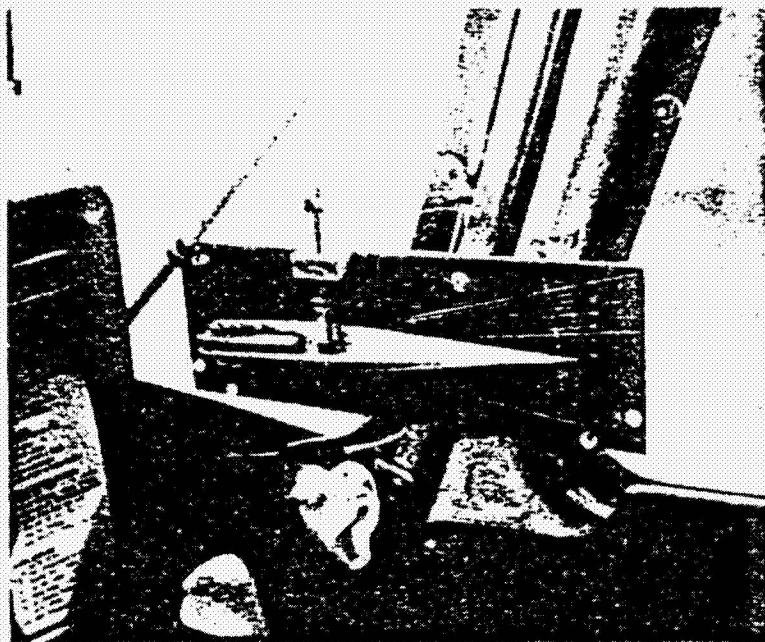


Figure 7.- Deck-angle inclinometer.

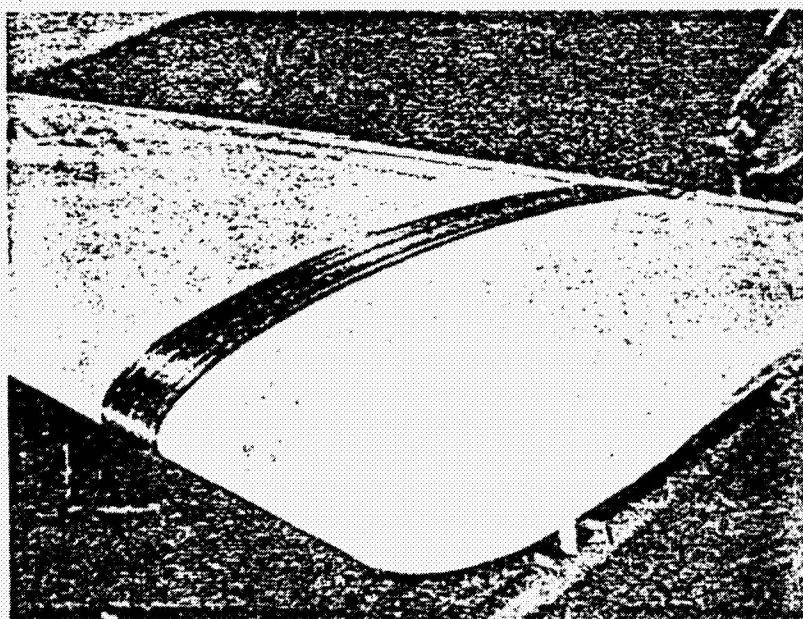


Figure 8.- Pressure-belt sensor.



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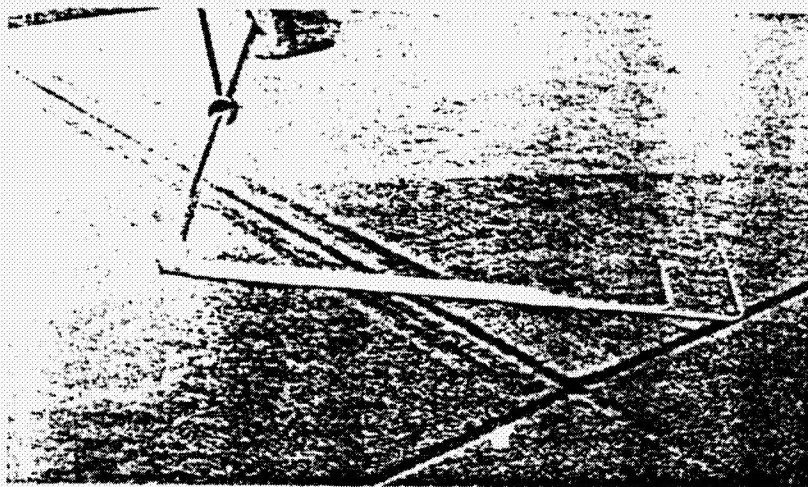


Figure 9.- Wake-survey probe.

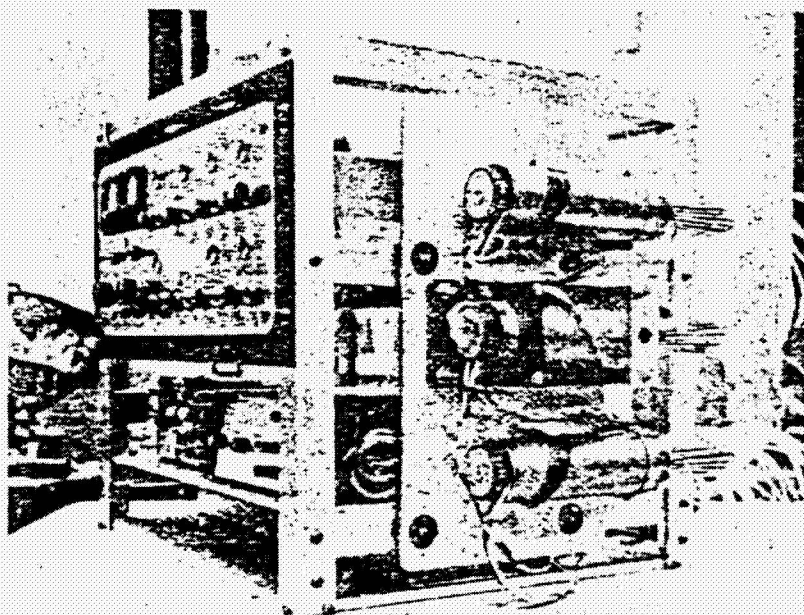


Figure 10.- Instrument rack.

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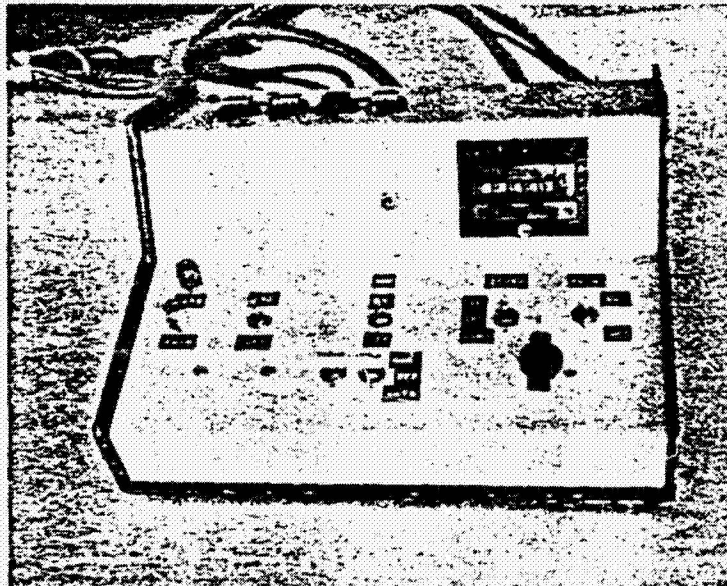


Figure 11.- Remote-control panel.

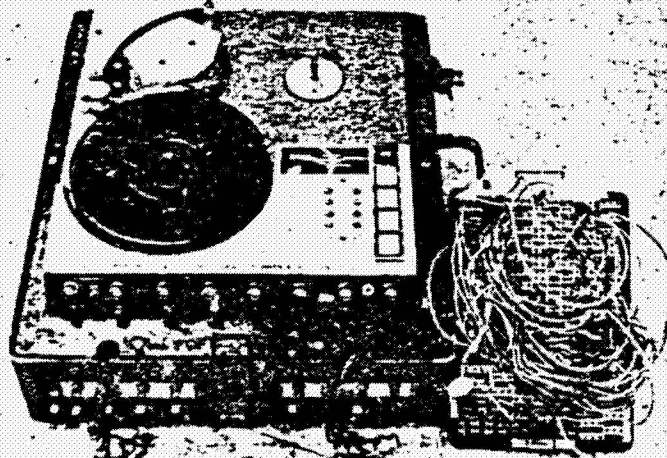


Figure 12.- FM analog recorder with patch panel.

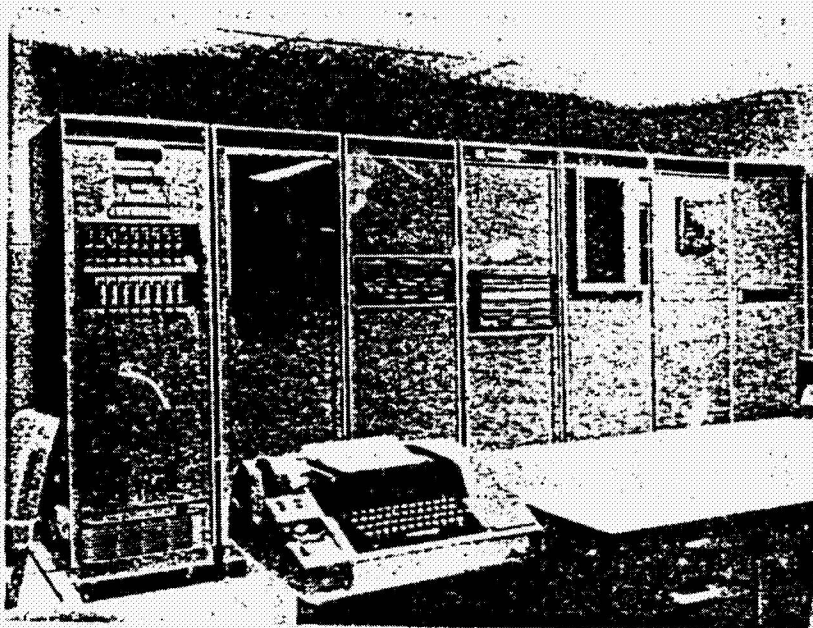


Figure 13.- Ground-based digital computer system.

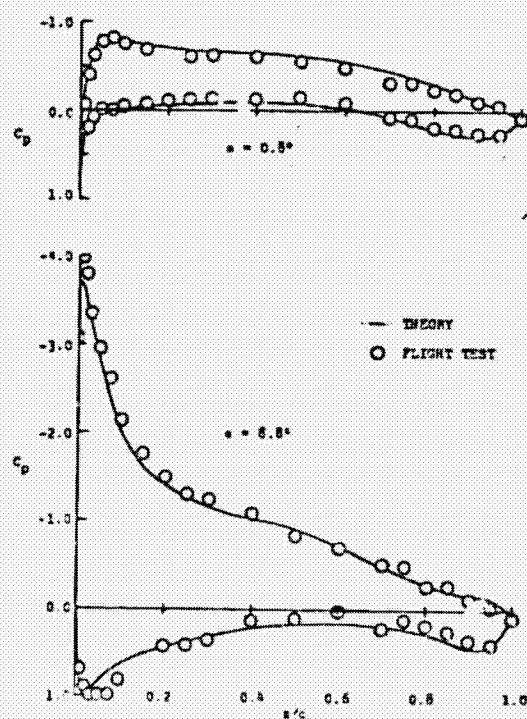


Figure 14.- Comparison of pressure distributions.

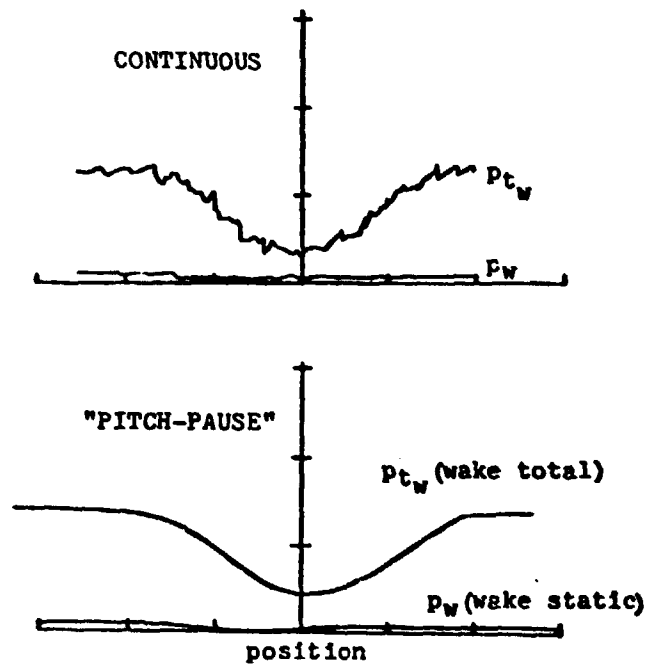


Figure 15.- Typical wake-survey profiles.

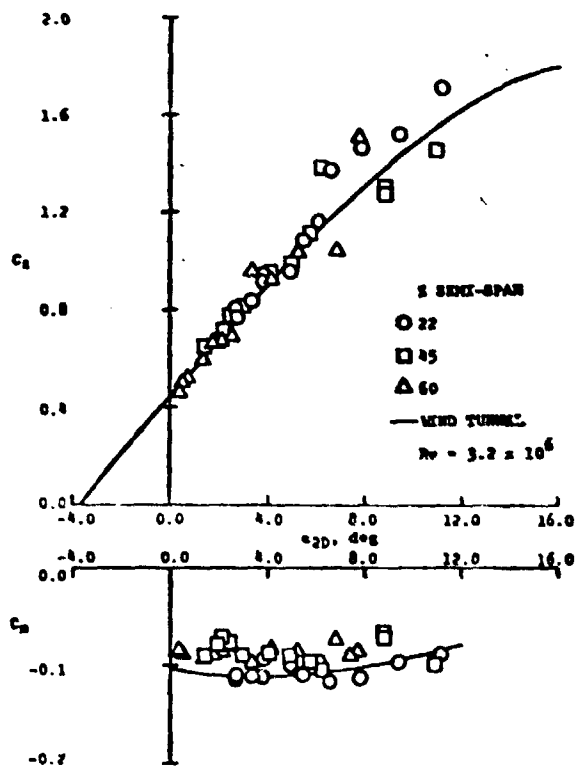


Figure 16.- Lift and moment coefficients.

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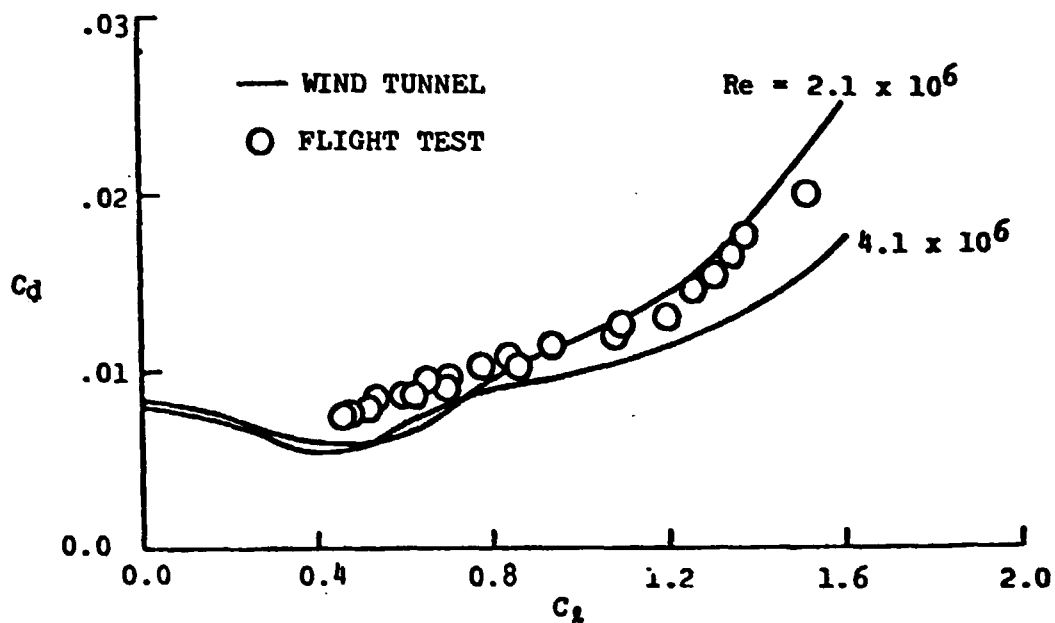


Figure 17.- Baseline drag polar.

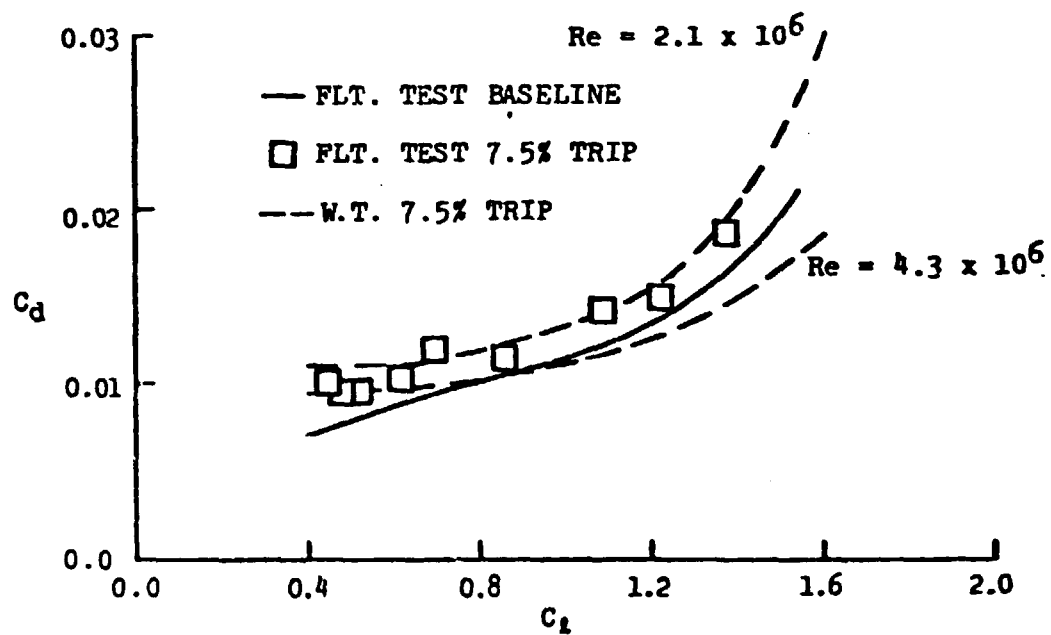


Figure 18.- 7.5% trip drag polar.

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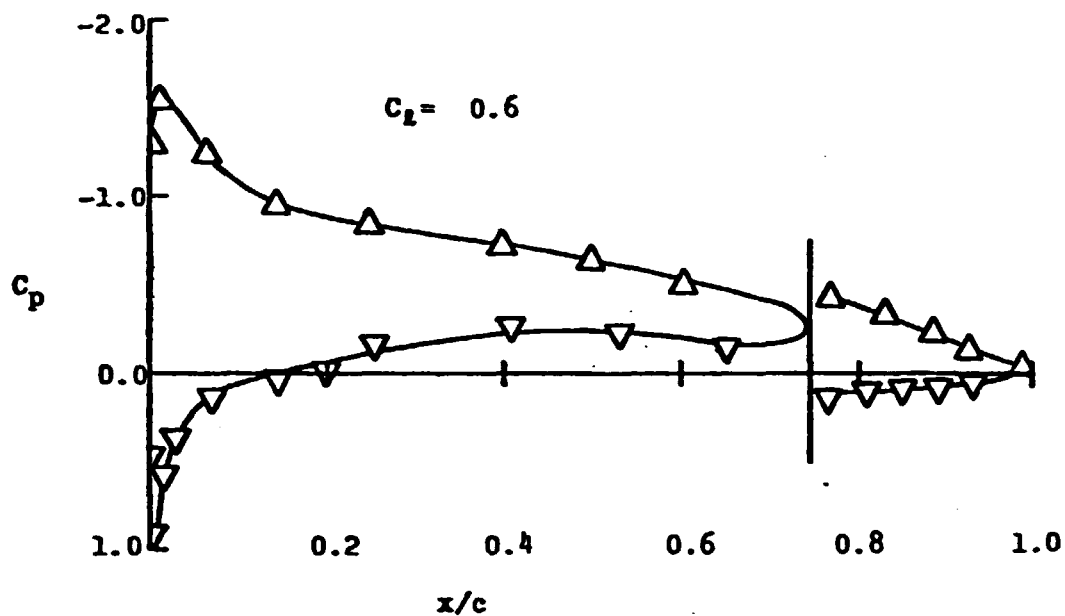


Figure 19.- Aileron-station pressure distribution.

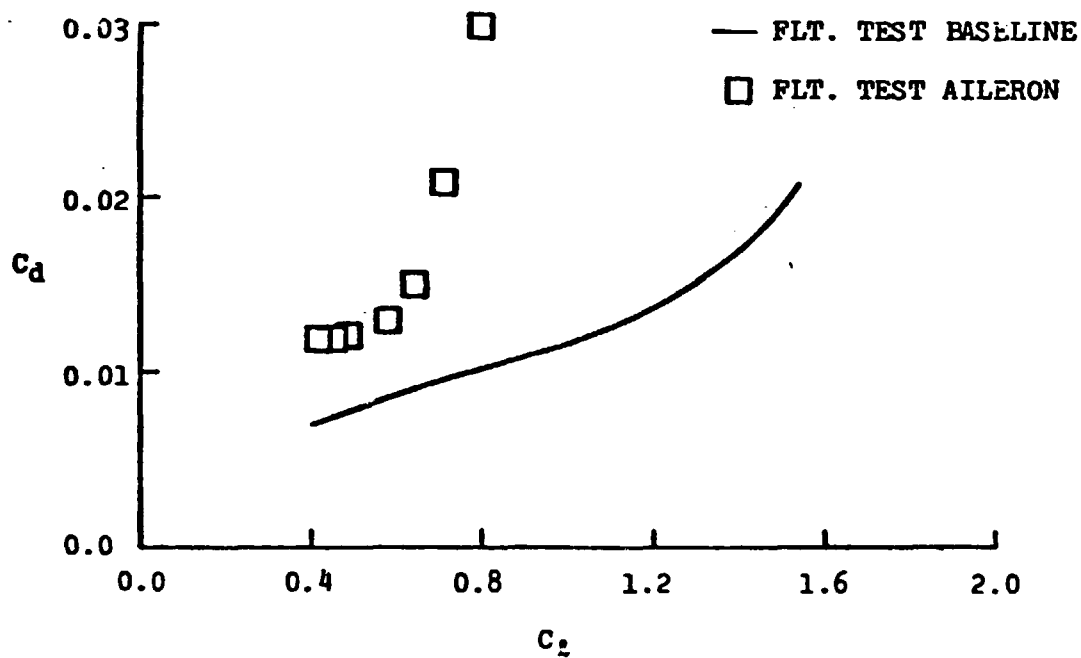


Figure 20.- Aileron-station drag polar.



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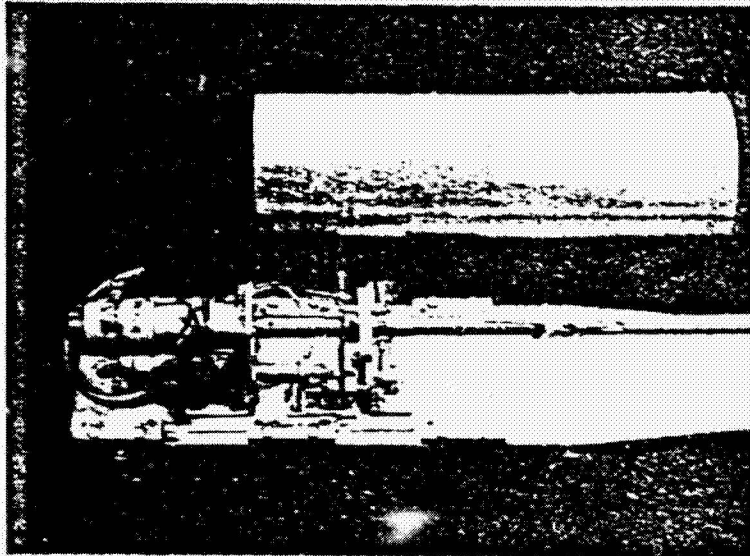


Figure 21.- New wake survey mechanism.

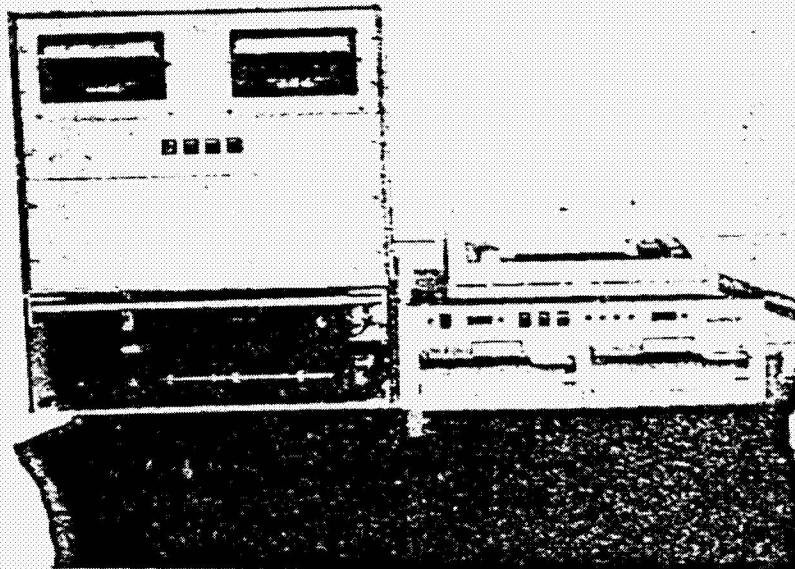


Figure 22.- Digital-data acquisition and reduction system.